

Design of a Manned Mars Mission using IMPRESS Technology

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Abstract

The objective of this creative investigation is the conceptual design of a propulsion system for a manned mission to Mars. Hydrogen and oxygen were selected as the propellants based upon their historical performance. The propellant storage system was a major consideration of the design due to the problems associated with storing hydrogen and oxygen. The most significant problems are the low density of hydrogen which requires large, heavy storage tanks and cryogenic storage of the propellants.

In order to circumvent these problems, new storage techniques have been incorporated into the design of the propulsion system. Water will be stored in high strength, low density graphite composite tanks until needed. The water will then be electrolyzed into hydrogen and oxygen and cryogenically stored in high strength, low density graphite epoxy composite tanks.

Incorporating these new technologies into the design significantly reduces the problems associated with using hydrogen and oxygen. Using the ideal rocket equation, a top level analysis was performed on five scenarios with different combinations of the presented technologies. Although initial calculations are positive, there are still limitations that must be overcome. Extensive testing and space qualification must be performed before these new technologies will be incorporated into a manned mission. More importantly, a manned spacecraft would be too heavy to launch from the surface of the Earth, therefore requiring that the spacecraft be constructed on orbit.

Table of Contents

Abstract.....	1
Table of Contents	2
Table of Figures.....	3
Background.....	4
Introduction	6
Specific Topics of Review.....	8
Mars Transfer Orbit	8
Hydrogen/Oxygen Rocket	13
IMPRESS	13
DOE/Ford T1000G Fuel Bladder	14
On Board Cryogenic Propellant Storage System	15
Top Level Design of a Liquid Rocket	15
Discussion of Limitation	21
Discussion of Application	22
Conclusion.....	23
References	24
Appendices	25

Table of Figures

Figure 1: Schematic of Hohmann Transfer	8
Figure 2: Schematic of One Tangent Burn.....	10
Figure 3: IMPRESS.....	13
Figure 4: Comparison of Tank Mass versus Technologies Used	23

Background

In 1957, the Soviet Union launched the world's first satellite Sputnik. This event started an era of intense space investigation, ultimately leading to man landing on the moon. With the continued exploration of the planet Mars a recurring question creeps into ones mind. Will man ever walk on Mars? Considering the curiosity of humans, the answer is most definitely yes. A manned mission to Mars is a complex and sensitive endeavor with two major questions, how and when. The spacecraft that is needed for such a journey would be so large that it would be impossible to launch from the Earth's surface. This leads one to conclude that the spacecraft would have to be constructed in space. Space construction of this magnitude would require a fully operational space station and extensive testing of spacecraft assembled on orbit. Therefore, the question of when such a mission could be accomplished becomes as dubious as how the mission can be accomplished.

Most theoretical manned Mars missions incorporate the use of nuclear technology to provide the necessary thrust levels for such an expedition. Nuclear power in space is an extremely sensitive issue that requires public education and political finesse. An excellent example of public concern about nuclear power in space is the Cassini mission to Saturn. Cassini uses radio-isotropic thermonuclear generators (RTG) which harness energy created by the natural decay of plutonium¹. An RTG is not a nuclear reactor with moving parts that can fail, but there was still extreme public concern about *nuclear* power on board the spacecraft.

Coupling the political factors surrounding nuclear power in space are the complexity of building a reactor in space and the untested reliability of a nuclear reactor in space.

¹ www.jpl.nasa.gov/cassini/rtg/power.htm

Theoretically a nuclear reactor should work in a space environment, but there have never been any prototypes launched or tests conducted to validate the use of nuclear power in space.

Considering all the negatives surrounding a nuclear powered spacecraft, it seems unlikely to design a mission that uses nuclear technology. Therefore, it is necessary to consider other technology in order to accomplish a manned mission within the next fifty years. A liquid engine that uses hydrogen and oxygen is able to provide a higher specific impulse (Isp) than any other existing technology. Isp is the ratio of the thrust to the weight flow rate of the propellant. In other words, Isp is a measure of the energy content of the propellants and how efficiently it is converted into thrust². The major drawback with using hydrogen and oxygen is long-term cryogenic storage. It would be extremely difficult to cryogenically store the propellants for the six month journey to Mars but impossible to cryogenically store the propellants for two and a half years until the return voyage.

Consider a scenario where the hydrogen and oxygen are stored as water until needed. The water is then separated into hydrogen and oxygen using electrolysis. The use of water eliminates the need for long-term cryogenic propellant storage. Once separated, the propellants will be cryogenically stored in high strength, graphite composite tanks to further reduce the mass of the system. Lawrence Livermore National Laboratory (LLNL) is designing and developing the technological advances that can help make this mission possible.

² Larson, W.J. and Wertz, J.R. p. 640.

Introduction

As we approach the twenty-first century, space exploration is continuing at a steady rate. Although NASA is moving toward smaller, less expensive missions, there is still the lingering idea of human exploration of our universe. With a multitude of robot exploration missions of Mars, it is only logical that the next advance will be an astronaut walking on Mars.

At this moment in time, human exploration of Mars is nearly impossible due to the duration of the voyage to and from Mars. In theory the only possible means of travel to Mars would require nuclear power. Political and environmental activists make nuclear travel almost impossible. This leads to new advances in other propulsion systems.

Liquid propulsion is the only other technology available at this time that is capable of producing enough specific impulse to send a manned mission to Mars. Hydrogen and Oxygen are the only propellant combination that a high enough Isp to accomplish such an endeavor. In order to store enough hydrogen and oxygen, cryogenics must be employed. Since the mission to Mars takes well over six months, with at least an eighteen month stay and a six month return journey, it is impossible to cryogenically store the propellants for the return mission.

Although long-term use of cryogenics is not feasible, imagine if water could be stored on board the spacecraft and converted to hydrogen and oxygen as needed. Thanks to Lawrence Livermore National Laboratory that dream is a reality. The Integrated Modular Propulsion and Regenerative Electro-Energy Storage System (IMPRESS) uses solar energy to create hydrogen and oxygen from water by electrolysis.

After the hydrogen and oxygen are created, they will be stored using cryogenics for a short duration until used. Cryogenics must be used in order to liquefy the propellants.

Without the cryogenic density, it becomes impossible to store the hydrogen and oxygen. The mass of the storage tanks becomes so great, that the mission is unfeasible. Titanium is presently the strongest, lightest material used for propellant storage tanks. Preliminary testing of graphite/epoxy composite tanks show a strength increase of eight fold compared to similar tanks created from Titanium³. Using the composite material further reduces the mass of the storage tanks.

Analysis of the Mars mission begins with a comparison of a Hohmann transfer and a one tangent burn. The transfer orbit was selected by comparing the time of flight with the additional ΔV needed. The ΔV was then used in a top-level analysis of the Mars mission. Different scenarios were considered and analyzed by using different combinations of IMPRESS, titanium tanks, the graphite/epoxy composite tanks and the on board cryogenic storage system. The scenarios were compared and the best one was selected.

³ Mitlitsky, Groot, Butler, and McElroy. p. 15.

Specific Topics of Review

Mars Transfer Orbit

The transfer orbit is the driving factor in determining the fuel needed. A Hohmann transfer was initially considered because of the efficiency, but the penalty is the time of flight. The Hohmann transfer requires approximately nine months transition time, which is too long for a manned mission. In order to reduce the time of flight, a one tangent burn was considered. By adjusting the angle to the transfer point, time of flight can be optimized versus the ΔV needed. Calculations for Hohmann transfer and One Tangent Burn comparisons are located in Appendix A.

Hohmann Transfer:

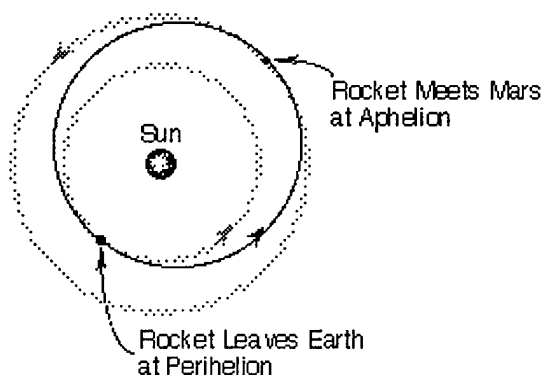


FIGURE 1: SCHEMATIC OF HOHMANN TRANSFER

In order to simplify calculations, the planetary orbits are assumed to be both circular and coplanar⁴. The energy of the transfer ellipse is calculated in order to determine the velocity of the transfer orbit at Earth and Mars.

⁴ Bate, R.R, Mueller, D.D., and White, J.E. p. 362.

Equation 1: Energy of Hohmann Transfer Ellipse

$$\varepsilon_T = \frac{-\mu_0}{r_1 + r_2} = \frac{-1}{1 + 1.524} = -0.3962 \frac{AU_{sun}^2}{TU_{sun}^2}$$

ε_T = Energy of Hohmann Transfer Ellipse

μ_0 = Gravitational Parameter of the sun

r_1 = Radius from Sun to Earth in Astronomical Units

r_2 = Radius from Sun to Mars in Astronomical Units

Equation 2: Velocity of Transfer Orbit at Earth/Mars

$$v_1 = \sqrt{2\left(\frac{\mu_{Sun}}{r_1} + \varepsilon_T\right)}$$

v_1 = Velocity of Transfer Orbit at Earth/Mars

r = Radius of Earth/Mars Orbit Around Sun

Equation 3: Velocity of Earth/Mars Orbit Around Sun

$$v_{planet} = \sqrt{\frac{\mu_{Sun}}{r_{planet}}}$$

v_{Planet} = Velocity of Earth/Mars Heliocentric Orbit

μ_{Sun} = Gravitational Parameter of Sun (1.3271544 E11 km³/sec²)

r_{Planet} = Radius of Earth/Mars Heliocentric Orbit

Equation 4 : Change in Velocity to get on/off Transfer Orbit

$$\Delta V_{on} = V_{tx} - V_{planet}$$

$$\Delta V_{off} = V_{planet} - V_{tx}$$

The velocity of the parking orbits at Earth and Mars are calculated using the gravitational parameters of the respective planet and the altitude of the parking orbit, as indicated in the following equation.

Equation 5: Velocity of Parking Orbit

$$v_{Park} = \sqrt{\frac{\mu_{Planet}}{r_{Park}}}$$

v_{Park} = Velocity of Parking Orbit

μ_{Planet} = Gravitational Parameter of Planet

r_{Park} = Radius of Parking Orbit

Following the patched conic method for interplanetary transfers, the energy of the transfer orbit at each planet is determined using the required change in velocity needed from equation 4 above. The energy at each planet is used to determine the velocity of the Hohmann transfer ellipse at perihelion and aphelion.

Equation 6: Energy of Hohmann Ellipse at perihelion and aphelion

$$\mathcal{E} = \frac{\Delta v_{planet}^2}{2}$$

Equation 7: Velocity of Hohmann Transfer Ellipse at perihelion and aphelion

$$v_h = \sqrt{2 \left(\frac{\mu_0}{r_{park}} + \mathcal{E} \right)}$$

Equation 8: Total Change in Velocity

$$\Delta v_1 = v_h - v_{park}$$

$$\Delta v_2 = v_{park} - v_h$$

$$\Delta v_{Total} = \Delta v_1 + \Delta v_2$$

Considering the affects of extended exposure to the space environment, the time of flight of the transfer orbit becomes the critical factor. The time of flight calculation for a Hohmann transfer from Earth to Mars is shown below.

Equation 9: Time of Flight

$$TOF = \pi \sqrt{\frac{(r_{earth} + r_{mars})^3}{8\mu_{sun}}} = \pi \sqrt{\frac{(2.524)^3}{8}} = 4.4512 TU_{sun} = 258.9 days$$

TOF = Time of Flight of Hohmann Transfer Orbit

One Tangent Burn:

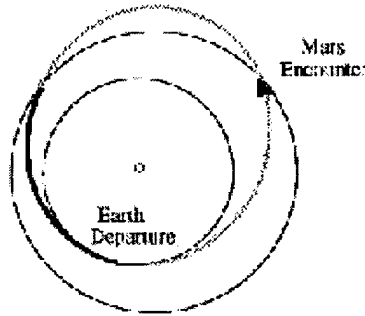


FIGURE 2: SCHEMATIC OF ONE TANGENT BURN

Due to the fact that time of flight is so critical when considering a manned mission, other transfer methods must be employed. The quickest transfer orbit is the one tangent burn

shown above. This transfer orbit consists of a perihelion burn which is tangent to the Earth's orbit about the sun and a non-tangential burn at the intersection with Mars. The angle to the transfer point can be adjusted to optimize the time of flight versus the additional change in velocity. Calculation of a one tangent burn commences with determining the eccentricity and semi-major axis of the transfer orbit as expressed in equations 10 through 12.

Equation 10: Inverse R ratio

$$R^{-1} = \frac{r_i}{r_f} = \frac{1}{1.524} = 0.6562$$

r_i = Radius from the Sun to the Earth
 r_f = Radius from the Sun to Mars

Equation 11: Eccentricity of Transfer Orbit

$$e_{trans} = \frac{R^{-1} - 1}{\cos(v_{transb}) - R^{-1}} (periapsis)$$

e_{trans} = Eccentricity of Transfer Orbit
 v_{trans} = Angle to Transfer Point

Equation 12: Semi-major Axis of Transfer Orbit

$$a_{trans} = \frac{r_{init}}{1 - e_{trans}} (periapsis)$$

a_{trans} = Semi-Major Axis of Transfer Orbit
 r_{init} = Radius from the Sun to the Earth

Similar to the Hohmann transfer, the velocity of Earth's orbit, Mar's orbit and the velocity of the transfer orbit had to be calculated in order to determine the change in velocity needed.

Equation 13: Velocity of Earth

$$v_{earth} = \sqrt{\frac{\mu_{sun}}{r_{earth:sun}}}$$

Equation 14: Velocity of Mars

$$v_{mars} = \sqrt{\frac{\mu_{sun}}{r_{mars:sun}}}$$

Equation 15: Velocity of Transfer Orbit at Earth

$$v_{trans_a} = \sqrt{\frac{2\mu}{r_{init}} - \frac{\mu}{a_{trans}}}$$

v_{trans_a} = Velocity of Transfer Orbit at Earth
 μ = Gravitational Parameter of the Sun in Astronomical Units

Equation 16: Velocity of Transfer Orbit at Mars

$$v_{trans_b} = \sqrt{\frac{2\mu}{r_{fin}} - \frac{\mu}{a_{trans}}}$$

v_{trans_b} = Velocity of Transfer Orbit at Mars

Equations 17 through 20 were used to calculate the ΔV at Earth, at Mars and the total change in velocity needed for the mission.

Equation 17: Change in Velocity at Earth

$$\Delta v_a = v_{trans_a} - v_{earth}$$

Equation 18: Flight Path Angle for Non-tangential Transfer

$$\phi_{trans_b} = \tan^{-1}\left(\frac{e_{trans} \sin(v_{trans_b})}{1 + e_{trans} \cos(v_{trans_b})}\right)$$

ϕ_{trans_b} = Flight Path Angle for Non-tangential Transfer

Equation 19: Change in Velocity at Mars

$$\Delta v_b = \sqrt{v_{trans_b}^2 + v_{fin}^2 - 2v_{trans_b} v_{fin} \cos(\phi_{trans_b})}$$

Equation 20: Total Change in Velocity for Mission

$$\Delta v_{otb} = |\Delta v_a| + |\Delta v_b|$$

The main reason for using the one tangent burn transfer orbit is to reduce the time of flight of the transfer orbit. The time of flight can be calculated using equations 21 and 22.

Equation 21: Eccentric Anomaly of Transfer Orbit at Transfer Point

$$E = \cos^{-1}\left(\frac{e_{trans} + \cos(v_{trans_b})}{1 + e_{trans} \cos(v_{trans_b})}\right)$$

E = Eccentric Anomaly of Transfer Orbit at Transfer Point

Equation 22: Time of Flight for One Tangent Burn

$$TOF_{trans} = \sqrt{\frac{a_{trans}^3}{\mu}} \{2k\pi + (E - e_{trans} \sin(E)) - (E_0 - e_{trans} \sin(E_0))\}$$

TOF = Time of Flight of One Tangent Burn Transfer Orbit

Comparison of the transfer angle versus the time of flight and required change in velocity concludes that a ϕ of 145 degrees produces a time of flight of 193 days. This is a 66 day savings on the time of flight with an increase in ΔV of only 723.5 meters per second. Hence the transfer orbit is much shorter with a very slight ΔV penalty.

Hydrogen/Oxygen Rocket

Based upon proven technology, hydrogen and oxygen produce the highest Isp of any existing technology. There are a few problems with the storage of hydrogen and oxygen, the size of the tanks and long-term cryogenic storage. Due to the low density of hydrogen, the storage tanks are large and heavy which can increase the spacecraft's inert mass. In order to account for the extra weight, more propellant is added, which increases the weight so more propellant is needed. The other problem is long-term cryogenic storage of the liquid oxygen and liquid hydrogen. This is the limiting factor in using oxygen and hydrogen for interplanetary missions. It would be complex to design the cryogenic system to last the six month journey to Mars. Yet it would be impossible to store the propellants for two and a half years for the return trip. Using new technology, discussed below, the disadvantages of using hydrogen and oxygen can be overcome.

IMPRESS

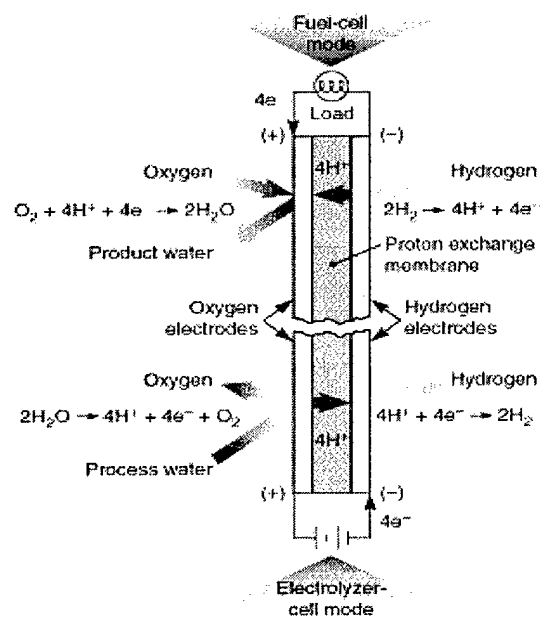


FIGURE 3: IMPRESS

Lawrence Livermore National Laboratory in Livermore, CA has created IMPRESS, which stands for Integrated Modular Propulsion and Regenerative Electro-energy Storage System. IMPRESS is an electrolyzer that uses electrical power to convert water into hydrogen and oxygen. In order to obtain an Isp of 435 seconds an oxidizer to fuel ratio (O/F) of 3.8 is desired, but IMPRESS creates an O/F between 7.5 - 9.1. The extra oxygen created can be used for life support for the manned mission. IMPRESS can also be used to create oxygen for breathing, the hydrogen created can be used for cold gas thruster attitude control. By using IMPRESS, the hydrogen and oxygen are stored on the spacecraft in the form of water eliminating any need for long-term cryogenic storage. After the water is converted into hydrogen and oxygen, cryogenics will be employed to further reduce the mass of the storage tanks.

DOE/Ford T1000G Fuel Bladder

Propellant mass is a major factor when considering the use of hydrogen and oxygen. Due to the low density of hydrogen, the storage tank is very large and heavy. Until recently, titanium was the only low density, high strength material that could be used for storage tanks. Lawrence Livermore National Laboratories has created a graphite composite tank called the T1000G. The T1000G has a proven performance factor of 50,800 meters compared to titanium's performance factor of 6350 meters, this is an eight fold increase in performance. By using T1000G technology, the storage tanks dimensions and mass are greatly reduced, thereby eliminating the other historical problem associated with using hydrogen as a fuel.

On Board Cryogenic Propellant Storage System⁵

Although long-term storage of the propellants is unfeasible, it is necessary to store the hydrogen and oxygen at cryogenic temperature for a short duration before use in order to reduce the size of the storage tanks. Without cryogenics, the densities of the hydrogen and oxygen gas require tanks that are extremely large. This increases the inert mass of the system making the mission impossible. Assuming it is possible to store the hydrogen and oxygen gas at 100 Kelvin without cryogenics, the mass of the storage tanks would be 1.45 million kilograms for the oxygen tank and 6.06 million kilograms for the hydrogen tank. If the spacecraft was outfitted with an 11,000 kilogram cryogenic storage unit⁶, the density obtained by using liquid hydrogen and liquid oxygen reduces the mass of the storage tanks to 830 kilograms for the oxygen tanks and 3500 kilograms for the hydrogen tank. Therefore, by adding the cryogenic storage system, the overall system mass is reduced significantly. Analysis was performed using the top-level design described below.

Top Level Design of a Liquid Rocket

After selecting the technology to aid in the mission, a top-level analysis was conducted to determine if this “water” rocket could work. Based upon the selected Isp and an estimated payload mass of 30,000 kilograms and an 11,000 kilogram cryogenic storage system, the ideal rocket equation was used to determine preliminary estimates for propellant, inert structure, initial and final mass.

⁵ Kohout,L.L.

⁶ Kohout,L.L. p. 7.

Equation 23: Mass of Propellant (ideal rocket equation)

$$m_{prop} = \frac{m_{pay} \left(e^{\frac{\Delta V}{I_{sp} * g_0}} - 1 \right) (1 - f_{inert})}{1 - f_{inert} * e^{\frac{\Delta V}{I_{sp} * g_0}}}$$

m_{prop} = mass of propellant

m_{pay} = mass of payload

f_{inert} = inert mass fraction

I_{sp} = Specific Impulse

g_0 = Gravity, 9.81 m/sec²

ΔV = Velocity Change

Equation 24: Mass of Inert Structure

$$m_{inert} = \frac{f_{inert}}{1 - f_{inert}} m_{prop}$$

m_{inert} = Mass of Inert Structure

Equation 25: Mass of Initial Spacecraft

$$m_{init} = m_{pay} + m_{inert} + m_{prop}$$

m_{init} = Mass of Initial Spacecraft

Equation 26: Mass of Final Spacecraft

$$m_{fin} = m_{pay} + m_{inert}$$

m_{fin} = Mass of Final Spacecraft

Before an in depth mission analysis was conducted, equations 23 through 26 were used to create Dummkopf Charts. These charts compare the I_{sp} required versus initial mass for different inert mass fractions based upon payload mass and ΔV required. The Dummkopf charts for this mission are located in Appendix B.

From figure B.12 in appendix B of Humble, Henry and Larson, an oxygen to fuel (O/F) ratio of 3.8 was selected based upon the 435 seconds of I_{sp} . Using figures B.9., B.10, and B.11. and an O/F of 3.8, chamber temperature (T_c), molecular weight (MW) and γ were determined. Engine length, diameter and mass were calculated using the following equations.

Equation 27: Thrust

$$F = \frac{F}{W} * g_0 * M_{init}$$

F = Thrust

F/W = Thrust to Weight Ratio

Equation 28: Mass of Engine

$$m_{engine} = \frac{F}{g_0 (25.2 \log F - 80.7)} \text{ kg}$$

m_{engine} = Mass of Engine

Equation 29: Length of Engine

$$L_{engine} = 0.00003042F + 327.7 \text{ cm}$$

L_{engine} = Length of Engine

Equation 30: Diameter of Engine

$$D_{engine} = 0.00002359F + 181.3 \text{ cm}$$

D_{engine} = Diameter of Engine

An expander cycle engine was chosen with regenerative cooling. The expander cycle engine was chosen because of the relative simplicity, low cost and high efficiency.

Regenerative cooling will be used in conjunction with the expander cycle. Cooling the thrust-chamber by heat transfer vaporizes the propellants before going into the chamber. In order to keep the system small and simple, the design incorporates a "blow down" concept. A "blow down" system uses extra propellant to maintain pressure in the tanks instead of using pumps or pressurants. This reduces both complexity and weight. Twenty percent extra is added to the propellant mass in order to keep the tanks pressurized.

Cryogenic storage of hydrogen and oxygen provide densities of 71 and 1142 kg/m³, respectively⁷. These densities do not apply when cryogenics are not being used. In order to determine the density of the hydrogen and oxygen gas at a given temperature and pressure the ideal gas law must be used. The volume of the storage tank was calculated using equations 31 and 32, listed below.

Equation 31: Density of Propellant

$$\rho = \frac{p}{R * Temp}$$

ρ = Density of Propellant

p = Tank Pressure

R = Specific Gas Constant

$Temp$ = Storage Temperature of Propellant

Equation 32: Volume of Propellant

$$V = \frac{m}{\rho}$$

V = Volume of Propellant

⁷ Humble, R.W., Henry, G.N., and Larson, W.J. p. 696.

After determining the pressurant requirements, the tank sizes and masses were calculated using the hoop stress and tank factor methods. By using the T1000G graphite composite tank, it was possible to obtain a tank factor (ϕ) of 50,800 meters. This is an eight fold increase over the next strongest material, titanium, with a tank factor of 6350 meters. The mass of the tanks was calculated with the subsequent equations.

Hoop Stress Method

Equation 33: Volume of Cylindrical Tank

$$V_c = \pi r_c^2 l_c$$

V_c = Volume of Cylindrical Tank

r_c = Radius of Cylindrical Tank

l_c = Length of Cylindrical Tank

Equation 34: Surface Area of Cylindrical Tank

$$A_c = 2\pi r_c l_c$$

A_c = Surface Area of Cylindrical Tanks

Equation 35: Thickness of Cylinder Wall

$$t_c = \frac{p_{burst} r_c}{F_{all}}$$

t_c = Thickness of Cylinder Wall

p_{burst} = Burst Pressure

F_{all} = Allowable Material Strength

Equation 36: Mass of Tank - Hoop Stress

$$m_{tank} = A_c t_c \rho_{mat}$$

m_{tank} = Mass of Tank

ρ_{mat} = Density of Tank Material

Tank Factor Method

Equation 37: Mass of Tank - Tank Factor

$$m_{tank} = \frac{p_{burst} * V_{tank}}{g_0 * \phi_{tank}}$$

ϕ_{tank} = Tank Factor

After sizing the tanks, the chamber and nozzle were sized. Columbium is a common material used for chambers and nozzles, therefore it was selected for this design. A bell nozzle was incorporated into the design in order to maximize efficiency. The chamber and nozzle were sized using the ensuing equations.

Equation 38: Exit Area

$$A_e = \frac{A_t}{M_e} \sqrt{\left[\left(\frac{2}{\gamma + 1} \right) \left(1 + \frac{\gamma - 1}{2} M_e^2 \right) \right]^{\frac{\gamma + 1}{\gamma - 1}}}$$

A_e = Exit Area

A_t = Area of the Throat

M_e = Exit Mach Number

γ = Isentropic Parameter

Equation 39: Length of Thrust Chamber

$$L_c = L \frac{A_t}{A_c}$$

L_c = Length of Thrust Chamber (m)

L = Chamber Characteristic Length (m)

Equation 40: Chamber Wall Thickness

$$t_w = f_s p \frac{r_c}{F_{tu}}$$

t_w = Thickness of Chamber Wall

f_s = Factor of Safety

p = Applied Pressure

r_c = Radius of Circular Cylinder

F_{tu} = Ultimate Tensile Strength

Equation 41: Mass of Thrust Chamber

$$m_{tc} = \pi \rho t_w \left(2r_{tc} L_{tc} + \frac{r_{tc}^2 - r_t^2}{\tan \theta_{tc}} \right)$$

m_{tc} = Mass of Thrust Chamber

ρ = Density of Wall Material

r_{tc} = Radius of Thrust Chamber

L_{tc} = Length of Thrust Chamber

r_t = Throat Radius

θ_{tc} = Constant Contraction Half Angle

Equation 42: Exit Diameter

$$D_e = \sqrt{\frac{4 \varepsilon A_t}{\pi}}$$

D_e = Exit Diameter

ε = Nozzle Expansion Ratio

Equation 43: Throat Diameter

$$D_t = 2 \sqrt{\frac{A_t}{\pi}}$$

D_t = Diameter of Throat

Equation 44: Nozzle Length

$$L_n = \frac{D_e - D_t}{2 \tan \theta_{cn}}$$

L_n = Length of Nozzle

θ_{cn} = Nozzle Cone Half Angle

Equation 45: Mass of Nozzle

$$m_n = \pi \rho t_w L_n (r_e + r_t)$$

m_n = Mass of Nozzle

r_e = Nozzle exit Radius

r_t = Throat Radius

After all the mass estimates were complete, a total inert mass and propellant mass was calculated. From this information, an inert mass fraction was determined and compared to the initial selected inert mass fraction. The initial inert mass fraction was adjusted until the two numbers converged.

Discussion of Limitation

The propulsion system of a manned mission to Mars is just one of a multitude of components that must be studied, designed and tested. Similarly, all aspects of an interplanetary manned mission are plagued with problems and limitations. The proposed propulsion system in this design is limited by the new technology presented.

By integrating IMPRESS, the T1000G graphite tanks and an on board cryogenic storage system into the propulsion system many problems associated with a hydrogen/oxygen system are eliminated. The problem is that these are new technologies that have not been space proven as of yet. Due to the complexity and sensitivity of a manned mission, extensive tests need to be conducted to assure that there will be no risk ensued by the astronauts. Although the preliminary ground tests have promising results, they may not pass the rigorous qualification tests in order to prove space worthy for a manned mission.

Aside from the untested technology presented, the mission is also limited by the realization that such a large spacecraft would need to be assembled on orbit. This in itself leads to a complex problem that requires a fully operational space station and extensive testing of on orbit assembly. This factor alone delays any possibility of a manned mission to mars that utilizes this or similar designs.

Discussion of Application

This conceptual design incorporates transfer orbit analysis comparing a Hohmann transfer with a one tangent burn. Using a top-level design, different combinations of the discussed technologies were integrated into the design and compared.

In order to reduce the time of flight, a one tangent burn with a flight path angle of 145 degrees was selected. This reduced the time of flight to 193 days while increasing the ΔV only 723.5 meters per second. For the top-level design, the ΔV is needed for the Earth escape burn and the Mars insertion burn. The Earth escape burn required a ΔV of 3290 meters per second while the Mars insertion burn needed 3067 meters per second of ΔV .

The payload of 30 tonne (30,000 kilograms) and an 11 tonne cryogenic storage system was used to determine the propellant needed for the Mars insertion burn. The propellant required for the Mars burn was added to the payload mass for the Earth escape burn. Using the equations 23 through 26, the inert, initial and final masses were determined for the spacecraft. Because there is no staging, the spacecraft will have a constant inert mass. Realizing this, the inert mass calculated for the Earth escape becomes the inert mass for the Mars insertion calculations. Iteration is necessary until the calculations converge.

Conclusion

Top level analysis of the mission indicates promising results. The use of IMPRESS and the T1000G graphite tanks reduces the overall system mass to workable levels. But, this is only made possible by employing an on board cryogenic storage system. The table below compares the tank masses based upon combinations of the new technologies.

Technology	Mass Hydrogen Tank (kg)	Mass Oxygen Tank (kg)
IMPRESS, T1000G, CSS	3,517	831
IMPRESS, T1000G	6,061,067	1,451,013
IMPRESS, CSS, Titanium	126,094	29,790
CSS, T1000G	5200	1228
CSS, Titanium	13,158,324	3,108,683

FIGURE 4: COMPARISON OF TANK MASS VERSUS TECHNOLOGIES USED

From Figure 4 above it is evident that without the incorporation of this or similar cutting edge technology, that a manned mission to Mars would be improbable using hydrogen and oxygen. Although this design does not conduct a mass breakdown of the specific components of the spacecraft, there is plenty of margin built into the top level design to account for unanticipated extras.

The primary objective of this creative investigation was to illustrate that a manned mission to Mars was possible using existing liquid rocket engine technology. Incorporation of state of the art and next generation storage techniques provide a reduction in the inert mass of the system that make the proposed mission feasible.

References

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*All equations were extracted from the following sources:

Hohmann Transfer Equations	Bate, Mueller and White
Patched Conic Method	Bate, Mueller and White
One Tangent Burn	Vallado and McClain
Top Level Mission Analysis	Humble, Henry and Larson

Appendix A: Orbit Calculations

Orbit transfer

LEO to Mars orbit, patched conic Hohmann



$$E_T = -\mu_0 / (r_1 + r_2) = -1 / (1 + 1.524) = -0.3962 \text{ AU}^2 / \text{TU}_0^2$$

$$v_T = \sqrt{2 \left(\frac{\mu_0}{r_1} + E_T \right)} = \sqrt{2 \left(\frac{1}{1} + -0.3962 \right)} = \sqrt{1.2076} = 1.0989 \text{ AU/TU}_0 = 32.73 \text{ km/sec}$$

• vel. of TE @ Earth

$$\text{T.O.F.} = \pi \sqrt{\frac{(r_1 + r_2)^3}{8\mu_0}} = \pi \sqrt{\frac{(2.524)^3}{8}} = 4.4512 \text{ TU}_0 = 0.788 \text{ yrs} = 258.9 \text{ days}$$

$$V_0 = \sqrt{\frac{\mu_s}{r_{s0}}} = \sqrt{\frac{1.32715 \text{ E11}}{1.496 \text{ E8}}} = 29.78 \text{ km/s}$$

• vel. of earth orbit around sun

$$V_M = \sqrt{\frac{\mu_s}{r_{sM}}} = \sqrt{\frac{1.32715 \text{ E11}}{2.278 \text{ E8}}} = 24.14 \text{ km/s}$$

• vel. of Mars orbit around sun

$$\Delta V_1 = V_{Tx} - V_0 = 2.94 \text{ km/s}$$

• ΔV needed @ Earth to get on trans. orbit

$$\Delta V_2 = V_M - \sqrt{\mu_s \left(\frac{2}{r_s} - \frac{1}{a_t} \right)} = 2.65 \text{ km/s}$$

• ΔV needed for Mars insertion

Leaving Earth

$$V_0 = \Delta V_0 = 2.94 \text{ km/s}$$

$$V_{\text{park}} = \sqrt{\frac{\mu_0}{r_{\text{park}}}} = \sqrt{\frac{3.986 \text{ E5}}{6878}} = 7.6127 \text{ km/s}$$

$$E = \frac{V_0^2}{2} = \frac{2.94^2}{2} = 4.322$$

$$V_{h0} = \sqrt{2 \left(\frac{\mu_0}{r_{\text{park}}} + E \right)} = \sqrt{2 \left(\frac{3.986 \text{ E5}}{6878} + 4.322 \right)} = 11.1602 \text{ km/s}$$

$$\Delta V_0 = 11.1602 - 7.6127 = 3.5475 \text{ km/s}$$

Entering Mars

$$V_{h0} = \Delta V_0 = 2.65 \text{ km/s}$$

$$V_{\text{park}} = \sqrt{\mu_M / r_{\text{park}}} = \sqrt{\frac{4.3079 \text{ E4}}{3893.16}} = 3.325 \text{ km/s}$$

$$E = V_{h0}^2 / 2 = 3.511$$

$$V_{h0} = \sqrt{2 \left(\frac{\mu_M}{r_{\text{park}}} + E \right)} = 5.3979 \text{ km/s}$$

$$\Delta V_M = 3.3125 - 5.3979 = -2.0854 \text{ km/s}$$

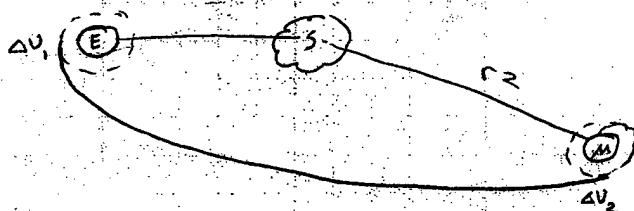
$$\Delta V_{\text{tot}} = 5.6329 \text{ km/s}$$

Orbit transfer

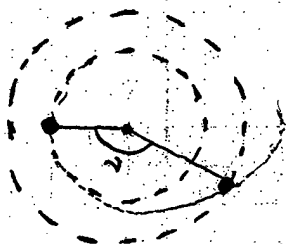
Leo to Mars - one-tangent burn, Patched Conic

$$r_1 = 1 \text{ AU}$$

$$r_2 = 1.524 \text{ AU}$$



$$\gamma = 145^\circ$$



$$R^{-1} = \frac{r_1}{r_2} = \frac{1}{1.524} = 0.6562$$

$$e_{\text{trans}} = \frac{R^{-1} - 1}{\cos(\gamma_{\text{trans}}) + R^{-1}} \quad (\text{periapsis})$$

$$q_{\text{trans}} = \frac{r_{\text{init}}}{1 + e_{\text{trans}}} \quad (\text{periapsis})$$

Plot shows optimal $\gamma_{\text{tx}} = 145^\circ$

$$e_{\text{tx}} = 0.233029$$

$$q_{\text{tx}} = 1.30383 \text{ AU}$$

$$\Delta V_a = 0.110418 \text{ AU/TU} = 3.28878 \text{ km/s}$$

$$\Delta V_p = 0.145319 \text{ AU/TU} = 4.3283 \text{ km/s}$$

$$\Delta V_{\text{tot}} = 0.255737 \text{ AU/TU} = 7.61709 \text{ km/s}$$

$$T_{\text{OT}} = 192 \text{ days}$$

Leaving Earth

$$v_{\text{eo}} = 3.28878 \text{ km/s}$$

$$v_{\text{park}} = \sqrt{\frac{\mu}{r_{\text{park}}}} = \sqrt{\frac{3.98665}{6878}} = 7.61268 \text{ km/s}$$

$$E = \frac{v_{\text{eo}}^2}{2} = 5.40804$$

$$v_{\text{ho}} = \sqrt{2 \left(\frac{\mu}{r_{\text{park}}} + E \right)} = 10.90262 \text{ km/s}$$

$$\Delta V_{\text{e}} = 10.90262 - 7.61268 = 3.28994 \text{ km/s}$$

entering Mars

$$v_{\text{eo}} = 4.3283 \text{ km/s}$$

$$v_{\text{parko}} = 3.325 \text{ km/s (Hohmann)}$$

$$E = \frac{v_{\text{eo}}^2}{2} = 9.36709 \text{ km/s}$$

$$v_{\text{ho}} = \sqrt{2 \left(\frac{\mu}{r_{\text{park}}} + E \right)} = 6.3915 \text{ km/s}$$

$$\Delta V_{\text{e}} = 3.325 - 6.3915 = -3.0665 \text{ km/s}$$

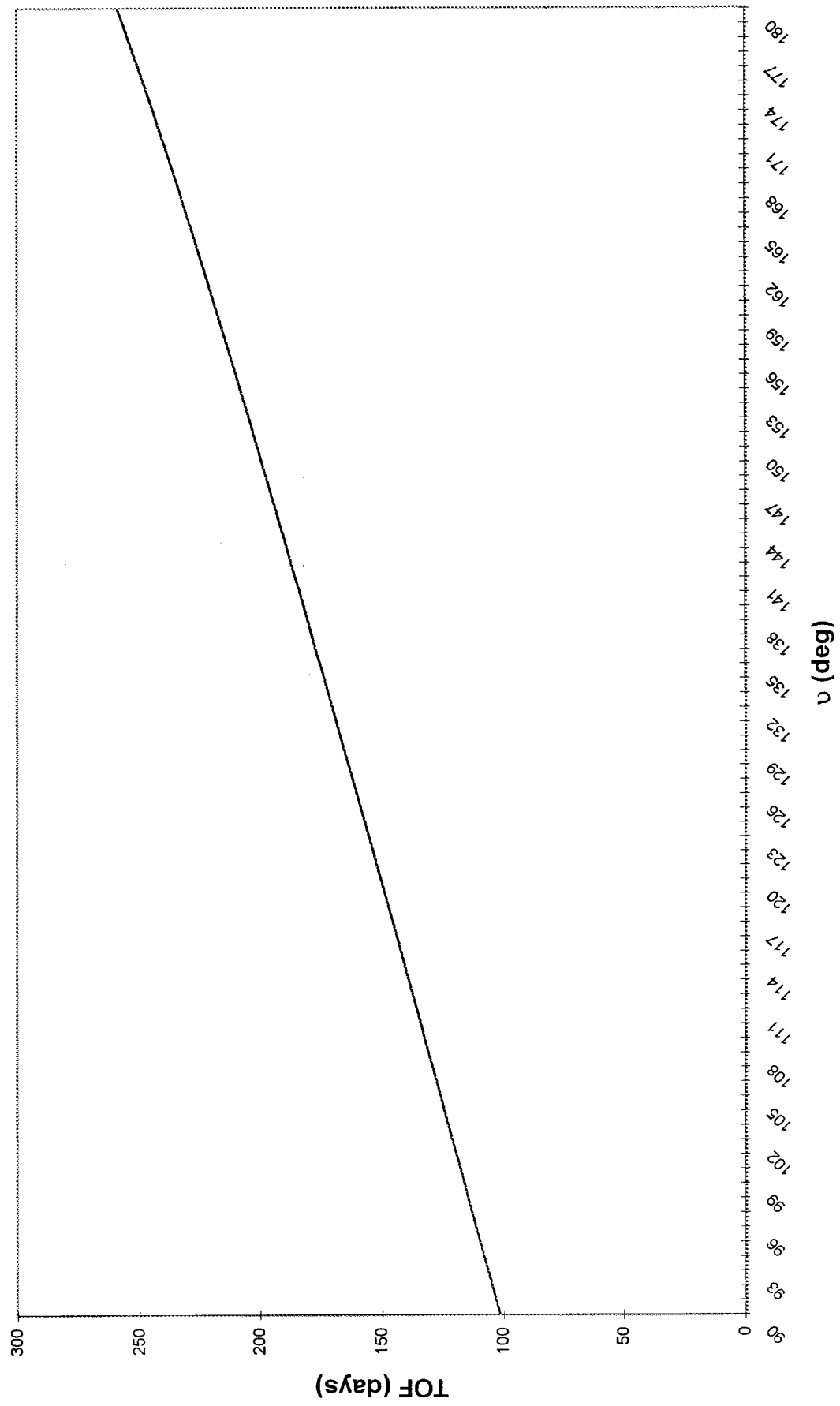
$$\Delta V_{\text{tot}} = 6.35644 \text{ km/s}$$

R-1 0.6562
 R init 1 AU
 R fin 1.524 AU
 μ 1
 V init 1 AU/TU
 V fin 0.810042 AU/TU

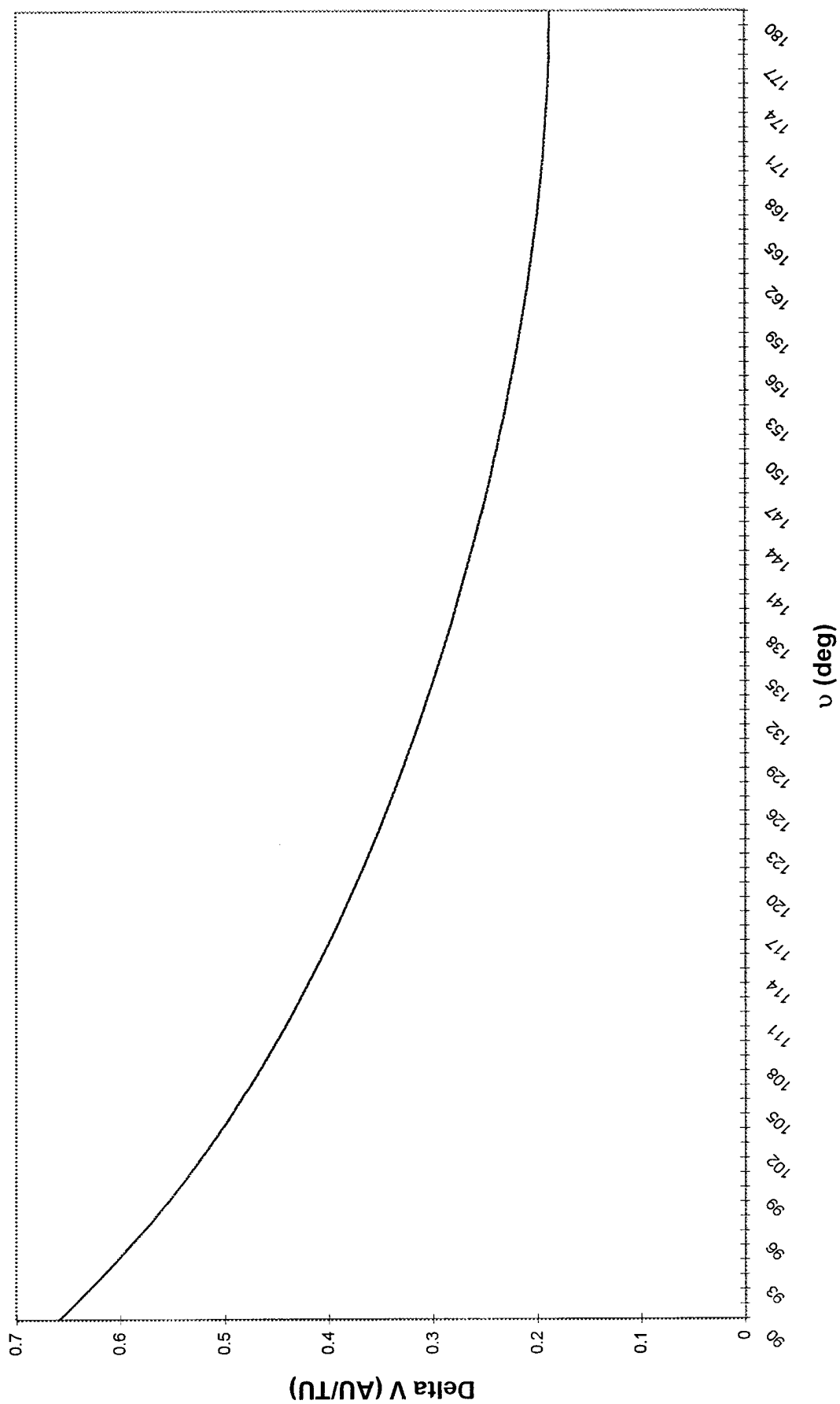
v_{txb}	v_{txb}	e trans	a trans	V_{txa}	V_{txb}	$\Delta V a$	ϕ_{txb}	$\Delta V b$	ΔV_{tot}	E	TOF (days)
90	1.570796	0.523926	2.100512	1.234474	0.914473	0.234474	0.482604	0.424396	0.65887	1.019343	101.4205602
91	1.58825	0.510352	2.042284	1.228964	0.907022	0.228964	0.475465	0.415207	0.644171	1.050278	103.0750919
92	1.605703	0.497468	1.989924	1.223711	0.899891	0.223711	0.468396	0.406322	0.630033	1.080665	104.7251722
93	1.623156	0.485226	1.9426	1.218698	0.893063	0.218698	0.461395	0.397724	0.616423	1.110549	106.3711664
94	1.640609	0.473582	1.899631	1.213912	0.88652	0.213912	0.454461	0.389397	0.603309	1.13997	108.0134213
95	1.658063	0.462497	1.860456	1.209338	0.880246	0.209338	0.447593	0.381325	0.590662	1.168962	109.6522661
96	1.675516	0.451935	1.824602	1.204963	0.874226	0.204963	0.440789	0.373494	0.578457	1.197559	111.2880146
97	1.692969	0.441863	1.791675	1.200776	0.868446	0.200776	0.434049	0.365892	0.566668	1.225787	112.920966
98	1.710423	0.43225	1.761339	1.196766	0.862894	0.196766	0.42737	0.358507	0.555274	1.253671	114.5514066
99	1.727876	0.423068	1.733308	1.192924	0.857557	0.192924	0.420752	0.351327	0.544252	1.281236	116.1796103
100	1.745329	0.414293	1.707337	1.18924	0.852425	0.18924	0.414193	0.344343	0.533583	1.3085	117.8058403
101	1.762783	0.405899	1.683215	1.185706	0.847487	0.185706	0.407693	0.337544	0.52325	1.335484	119.4303492
102	1.780236	0.397865	1.660758	1.182314	0.842734	0.182314	0.401249	0.330922	0.513236	1.362203	121.0533808
103	1.797689	0.390171	1.639805	1.179055	0.838157	0.179055	0.394861	0.324469	0.503524	1.388674	122.6751701
104	1.815142	0.382799	1.620217	1.175925	0.833747	0.175925	0.388528	0.318176	0.494101	1.414911	124.2959444
105	1.832596	0.37573	1.601871	1.172915	0.829497	0.172915	0.382249	0.312037	0.484952	1.440927	125.9159237
106	1.850049	0.368949	1.584657	1.170021	0.8254	0.170021	0.376021	0.306044	0.476065	1.466734	127.5353216
107	1.867502	0.36244	1.568479	1.167236	0.821447	0.167236	0.369845	0.300192	0.467428	1.492343	129.1543457
108	1.884956	0.356189	1.553252	1.164555	0.817634	0.164555	0.363719	0.294474	0.459029	1.517765	130.7731981
109	1.902409	0.350185	1.538898	1.161974	0.813954	0.161974	0.357641	0.288884	0.450859	1.543009	132.3920758
110	1.919862	0.344413	1.525351	1.159488	0.8104	0.159488	0.351612	0.283419	0.442907	1.568084	134.0111714
111	1.937315	0.338863	1.512547	1.157093	0.806969	0.157093	0.34563	0.278071	0.435164	1.592999	135.6306735
112	1.954769	0.333525	1.500432	1.154784	0.803655	0.154784	0.339694	0.272838	0.427622	1.617761	137.2507666
113	1.972222	0.328388	1.488956	1.152557	0.800453	0.152557	0.333802	0.267714	0.420272	1.642377	138.8716323
114	1.989675	0.323444	1.478073	1.15041	0.797358	0.15041	0.327955	0.262696	0.413106	1.666855	140.4934488
115	2.007129	0.318682	1.467743	1.148339	0.794366	0.148339	0.32215	0.257779	0.406118	1.6912	142.1163918
116	2.024582	0.314096	1.457929	1.14634	0.791474	0.14634	0.316388	0.25296	0.3993	1.715419	143.7406348
117	2.042035	0.309677	1.448596	1.144411	0.788678	0.144411	0.310666	0.248235	0.392647	1.739518	145.3663489
118	2.059489	0.305418	1.439714	1.142549	0.785973	0.142549	0.304985	0.243602	0.386151	1.763501	146.9937037
119	2.076942	0.301312	1.431254	1.140751	0.783357	0.140751	0.299342	0.239057	0.379808	1.787374	148.6228671
120	2.094395	0.297353	1.423191	1.139014	0.780826	0.139014	0.293738	0.234597	0.373612	1.811141	150.2540059
121	2.111848	0.293536	1.415499	1.137337	0.778377	0.137337	0.288172	0.23022	0.367557	1.834807	151.8872856
122	2.129302	0.289853	1.408159	1.135717	0.776008	0.135717	0.282642	0.225923	0.36164	1.858376	153.5228711
123	2.146755	0.2863	1.401149	1.134152	0.773716	0.134152	0.277147	0.221704	0.355856	1.881852	155.160927
124	2.164208	0.282871	1.39445	1.132639	0.771497	0.132639	0.271688	0.21756	0.3502	1.905239	156.801617
125	2.181662	0.279563	1.388046	1.131178	0.76935	0.131178	0.266262	0.21349	0.344668	1.928541	158.4451052
126	2.199115	0.27637	1.381921	1.129765	0.767272	0.129765	0.26087	0.209491	0.339257	1.95176	160.0915554
127	2.216568	0.273288	1.37606	1.1284	0.76526	0.1284	0.25551	0.205562	0.333962	1.974901	161.7411321
128	2.234021	0.270312	1.37045	1.127081	0.763314	0.127081	0.250182	0.2017	0.328782	1.997966	163.3939998
129	2.251475	0.26744	1.365077	1.125807	0.76143	0.125807	0.244884	0.197905	0.323712	2.020959	165.0503241
130	2.268928	0.264668	1.359929	1.124574	0.759608	0.124574	0.239617	0.194175	0.318749	2.043882	166.7102713
131	2.286381	0.261991	1.354997	1.123384	0.757844	0.123384	0.234378	0.190508	0.313892	2.066737	168.3740087
132	2.303835	0.259407	1.350269	1.122233	0.756137	0.122233	0.229169	0.186903	0.309136	2.089529	170.0417051
133	2.321288	0.256913	1.345737	1.121121	0.754486	0.121121	0.223987	0.18336	0.304481	2.112258	171.7135304
134	2.338741	0.254505	1.34139	1.120047	0.752888	0.120047	0.218832	0.179876	0.299923	2.134928	173.3896564
135	2.356194	0.252181	1.337222	1.119009	0.751343	0.119009	0.213703	0.176451	0.29546	2.15754	175.0702566
136	2.373648	0.249938	1.333224	1.118006	0.749849	0.118006	0.2086	0.173085	0.291091	2.180098	176.7555067
137	2.391101	0.247774	1.329388	1.117038	0.748405	0.117038	0.203522	0.169776	0.286814	2.202602	178.4455842
138	2.408554	0.245686	1.325709	1.116103	0.747009	0.116103	0.198469	0.166524	0.282627	2.225056	180.1406693
139	2.426008	0.243673	1.322179	1.115201	0.74566	0.115201	0.193438	0.163328	0.278529	2.24746	181.8409446
140	2.443461	0.241731	1.318792	1.11433	0.744356	0.11433	0.188431	0.160188	0.274518	2.269818	183.5465956

141	2.460914	0.239858	1.315544	1.113489	0.743098	0.113489	0.183446	0.157104	0.270593	2.29213	185.2578108
142	2.478368	0.238054	1.312429	1.112679	0.741883	0.112679	0.178482	0.154075	0.266753	2.314399	186.9747815
143	2.495821	0.236315	1.309441	1.111897	0.74071	0.111897	0.17354	0.151101	0.262998	2.336626	188.6977028
144	2.513274	0.234641	1.306576	1.111144	0.739579	0.111144	0.168617	0.148182	0.259326	2.358813	190.4267731
145	2.530727	0.233029	1.30383	1.110418	0.738488	0.110418	0.163715	0.145319	0.255737	2.380962	192.1621946
146	2.548181	0.231478	1.301199	1.10972	0.737438	0.10972	0.158831	0.142511	0.252231	2.403073	193.9041737
147	2.565634	0.229986	1.298678	1.109048	0.736425	0.109048	0.153966	0.139759	0.248807	2.425149	195.6529207
148	2.583087	0.228553	1.296265	1.108401	0.735451	0.108401	0.149118	0.137063	0.245464	2.447191	197.4086505
149	2.600541	0.227176	1.293955	1.10778	0.734514	0.10778	0.144288	0.134424	0.242204	2.4692	199.1715827
150	2.617994	0.225854	1.291745	1.107183	0.733614	0.107183	0.139474	0.131843	0.239026	2.491178	200.9419418
151	2.635447	0.224586	1.289633	1.10661	0.732749	0.10661	0.134676	0.12932	0.23593	2.513126	202.7199575
152	2.6529	0.22337	1.287615	1.106061	0.73192	0.106061	0.129894	0.126856	0.232917	2.535045	204.5058649
153	2.670354	0.222207	1.285689	1.105535	0.731124	0.105535	0.125126	0.124452	0.229987	2.556937	206.2999046
154	2.687807	0.221094	1.283852	1.105031	0.730363	0.105031	0.120373	0.12211	0.227141	2.578802	208.1023235
155	2.70526	0.220031	1.282102	1.10455	0.729635	0.10455	0.115633	0.11983	0.22438	2.600643	209.9133745
156	2.722714	0.219016	1.280437	1.104091	0.728939	0.104091	0.110907	0.117615	0.221705	2.622246	211.7333172
157	2.740167	0.21805	1.278854	1.103653	0.728276	0.103653	0.106193	0.115465	0.219118	2.644254	213.562418
158	2.75762	0.21713	1.277351	1.103236	0.727644	0.103236	0.101491	0.113382	0.216618	2.666026	215.4009505
159	2.775074	0.216256	1.275927	1.10284	0.727043	0.10284	0.0968	0.111369	0.214209	2.687778	217.2491958
160	2.792527	0.215428	1.27458	1.102465	0.726474	0.102465	0.092121	0.109426	0.211891	2.709511	219.1074428
161	2.80998	0.214644	1.273309	1.102109	0.725934	0.102109	0.087451	0.107557	0.209666	2.731225	220.9759887
162	2.827433	0.213905	1.272111	1.101774	0.725425	0.101774	0.082792	0.105762	0.207536	2.752922	222.8551394
163	2.844887	0.213209	1.270985	1.101458	0.724945	0.101458	0.078142	0.104046	0.205503	2.774603	224.7452096
164	2.86234	0.212555	1.26993	1.101161	0.724494	0.101161	0.0735	0.102409	0.203569	2.796268	226.6465236
165	2.879793	0.211944	1.268946	1.100883	0.724072	0.100883	0.068867	0.100854	0.201737	2.817919	228.5594155
166	2.897247	0.211375	1.268029	1.100625	0.723679	0.100625	0.064241	0.099383	0.200008	2.839557	230.4842295
167	2.9147	0.210847	1.267181	1.100385	0.723314	0.100385	0.059623	0.098	0.198385	2.861182	232.421321
168	2.932153	0.210359	1.266399	1.100163	0.722977	0.100163	0.055011	0.096707	0.19687	2.882796	234.3710563
169	2.949606	0.209912	1.265682	1.09996	0.722667	0.09996	0.050406	0.095506	0.195466	2.904399	236.3338137
170	2.96706	0.209505	1.265031	1.099775	0.722386	0.099775	0.045805	0.094399	0.194174	2.925992	238.3099838
171	2.984513	0.209138	1.264444	1.099608	0.722132	0.099608	0.04121	0.093389	0.192997	2.947577	240.2999702
172	3.001966	0.208811	1.26392	1.099459	0.721905	0.099459	0.03662	0.092478	0.191938	2.969154	242.3041898
173	3.01942	0.208522	1.263459	1.099328	0.721705	0.099328	0.032034	0.091669	0.190997	2.990724	244.323074
174	3.036873	0.208273	1.263061	1.099214	0.721532	0.099214	0.027451	0.090964	0.190178	3.012288	246.3570689
175	3.054326	0.208062	1.262725	1.099119	0.721386	0.099119	0.022871	0.090363	0.189482	3.033847	248.406636
176	3.071779	0.207889	1.26245	1.09904	0.721266	0.09904	0.018294	0.08987	0.18891	3.055401	250.4722535
177	3.089233	0.207756	1.262237	1.098979	0.721174	0.098979	0.013719	0.089485	0.188464	3.076952	252.5544161
178	3.106686	0.20766	1.262084	1.098936	0.721107	0.098936	0.009145	0.089209	0.188145	3.0985	254.6536371
179	3.124139	0.207603	1.261993	1.09891	0.721068	0.09891	0.004572	0.089043	0.187953	3.120047	256.7704479
180	3.141593	0.207584	1.261963	1.098901	0.721054	0.098901	3.21E-17	0.088987	0.187889	3.141593	258.9054002

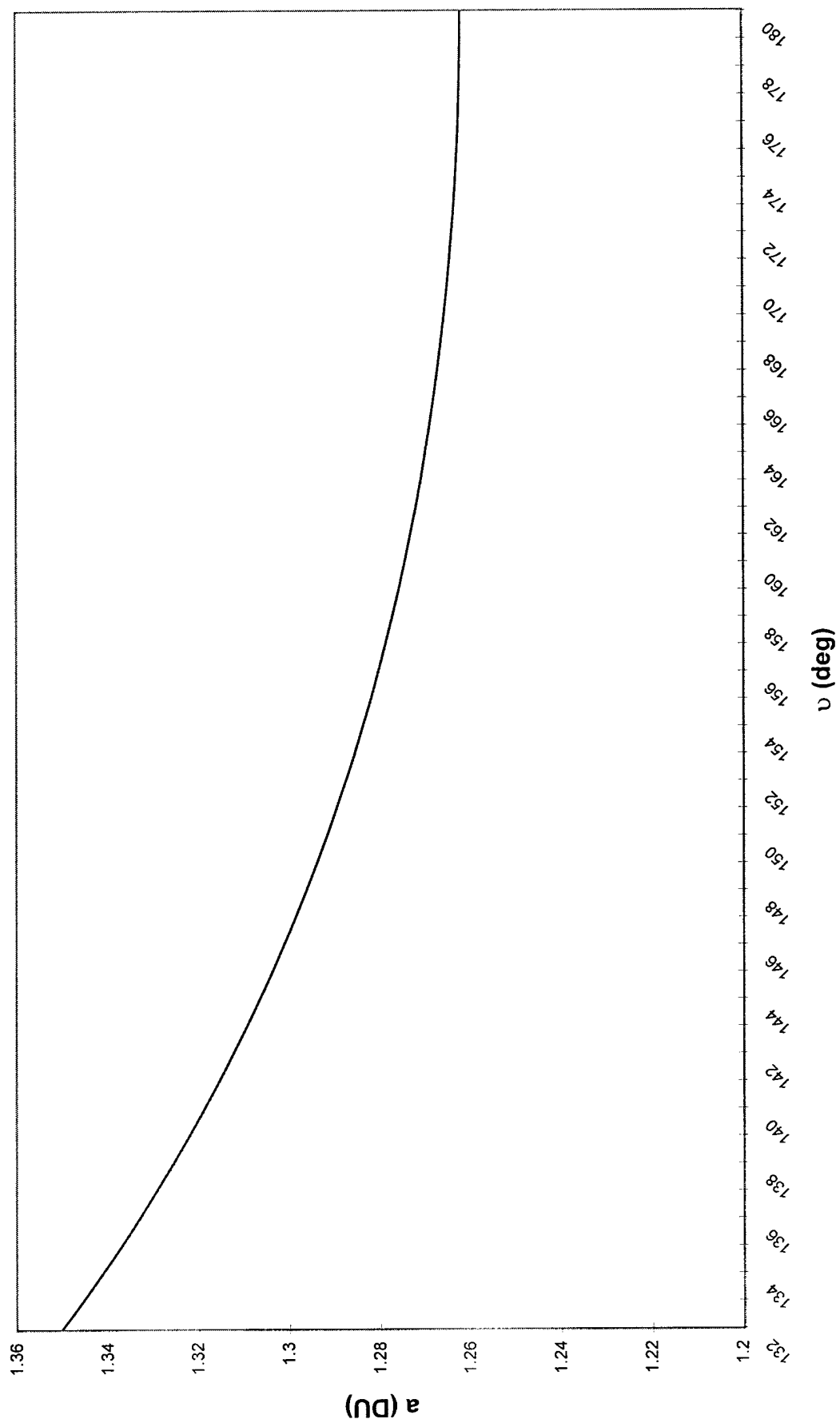
Time of Flight vs. ν



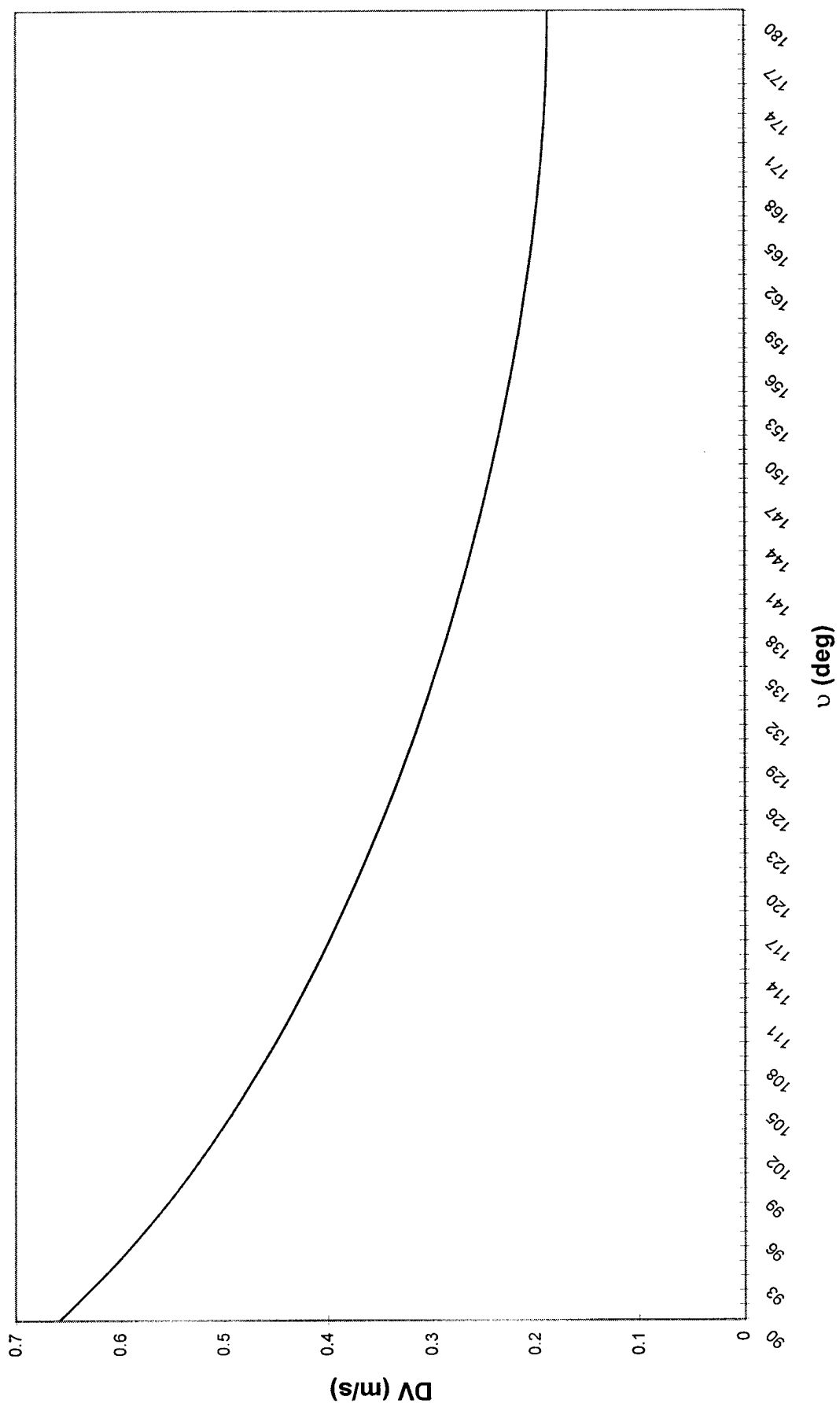
Delta V vs. ν



ν vs. a of transfer orbit



ν vs. DV



Appendix B: Preliminary Mission Analysis using Dummkopf Charts

Information to determine the Dumpskip chart for Earth Escape

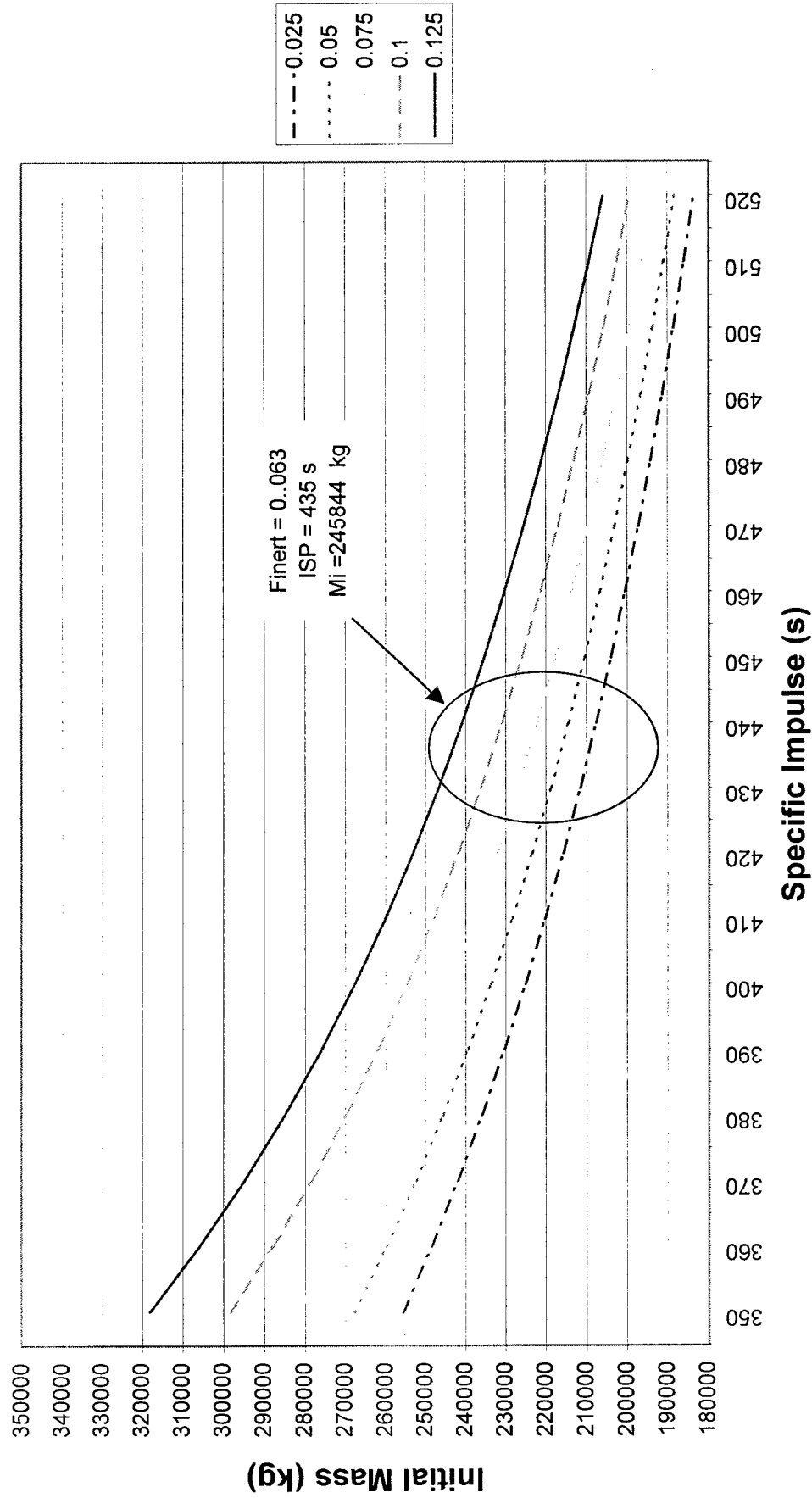
M payload 84046 kg
dV 3390 m/s
gravity 9.81 m/s²

f inert ISP	0.025					0.05					0.075					0.1					0.125									
	Mprop	Minert	MI	MF	Min ISP	Feasible	Mprop	Minert	MI	MF	Min ISP	Feasible	Mprop	Minert	MI	MF	Min ISP	Feasible	Mprop	Minert	MI	MF	Min ISP	Feasible	Mprop	Minert	MI	MF	Min ISP	Feasible
350	157628.5	4041.759	252716.3	9607.76	90.91435	Yes	165097.2	8689.327	297832.5	102715.3	111.9469	Yes	173761.1	14008.96	348102.1	108136	129.4739	Yes	181584.4	26024.06	359474.1	122020.1	181.2739	Yes	190154.9	45855.6	359474.1	122020.1	181.2739	Yes
360	150638.1	3802.464	248544.5	96067.48		Yes	157618.8	8235.51	292403.7	101863.9		Yes	165865.2	13779.4	348102.1	106254.4		Yes	175814.9	25116.41	348102.1	119162.4		Yes	185454.9	45855.6	348102.1	119162.4		Yes
370	144313.3	3891.778	241895.1	97435.78		Yes	150439.8	7817.883	252403.7	101863.9		Yes	157618.8	12779.4	348102.1	106254.4		Yes	165865.2	18460.58	348102.1	119162.4		Yes	175814.9	25116.41	348102.1	119162.4		Yes
380	138295.2	3465.031	235887.2	97592.03		Yes	144011	7578.525	245638.5	101625.5		Yes	144109.8	11208.42	348102.1	106254.4		Yes	151081.1	16784.57	348102.1	119162.4		Yes	160987.3	23871.04	348102.1	119162.4		Yes
390	132626	3546.031	230877.8	97521.03		Yes	138017	7859.833	233646.2	101313.9		Yes	144109.8	11208.42	348102.1	106254.4		Yes	151081.1	16784.57	348102.1	119162.4		Yes	160987.3	23871.04	348102.1	119162.4		Yes
400	127575.6	3275.637	225078.5	97321.84		Yes	132550.2	6980.012	233646.2	101028		Yes	138183.1	11202.42	348102.1	105748.4		Yes	144539.8	18258.86	348102.1	119162.4		Yes	151552.7	21707.88	348102.1	119162.4		Yes
410	123648.8	3155.087	220249.9	97201.11		Yes	127582.6	6435.238	22353.5	100511.8		Yes	132550.2	6980.012	233646.2	101028		Yes	144539.8	18258.86	348102.1	119162.4		Yes	151552.7	21707.88	348102.1	119162.4		Yes
420	115715.6	2957.175	207475.5	96855.5		Yes	118465	6235.001	218748	100281		Yes	127582.6	6435.238	22353.5	100511.8		Yes	132550.2	6980.012	348102.1	119162.4		Yes	144539.8	18258.86	348102.1	119162.4		Yes
430	114569.8	2937.887	211552.5	96885.5		Yes	118465	6235.001	218748	100281		Yes	127582.6	6435.238	22353.5	100511.8		Yes	132550.2	6980.012	348102.1	119162.4		Yes	144539.8	18258.86	348102.1	119162.4		Yes
440	110740.7	2838.504	207626.2	96885.5		Yes	114375.7	6019.775	214441.5	100058		Yes	118474.9	5608.071	222177	103652.1		Yes	123133.1	13681.45	348102.1	119162.4		Yes	128473	18352.26	348102.1	119162.4		Yes
450	107151.6	2747.478	203945.1	96792.48		Yes	110551.3	5818.488	210415.9	99864.49		Yes	114378.3	5273.754	217896.1	103181.8		Yes	118711.9	13190.21	348102.1	119162.4		Yes	123607.5	17668.78	348102.1	119162.4		Yes
460	103781.3	2661.059	200483.3	96707.06		Yes	108607.3	5629.855	206043.1	99675.86		Yes	110544.3	4962.049	213533.3	103599		Yes	115151.5	12525.1	348102.1	119162.4		Yes	120519.8	16433.7	348102.1	119162.4		Yes
470	100511.8	2575.555	197058.5	96584.4		Yes	105443.3	5462.049	202533.3	99359.9		Yes	105941.4	4871.861	209672	102718		Yes	110705.9	12302.99	348102.1	119162.4		Yes	118505.9	16433.7	348102.1	119162.4		Yes
480	97027.8	2503.148	194171.9	96548.15		Yes	100436.7	5286.143	198708.9	99321.14		Yes	103583.9	4848.693	206028.6	102444.7		Yes	107127.2	11802.75	348102.1	119162.4		Yes	111148.4	15878.08	348102.1	119162.4		Yes
490	94027.8	2450.838	191279.5	96478.15		Yes	97454.18	5129.167	196029.3	99175.17		Yes	100414.4	4811.712	209022.2	102187.7		Yes	103740.7	11528.75	348102.1	119162.4		Yes	107505.4	14357.02	348102.1	119162.4		Yes
500	92136.82	2382.833	188545.3	96408.48		Yes	94639.34	4981.018	193686.4	99027.02		Yes	97458.62	4981.018	193686.4	101845.6		Yes	97538.45	10838.83	348102.1	119162.4		Yes	104090.2	14357.02	348102.1	119162.4		Yes
510	88613.15	2287.773	185959.9	96343.77		Yes	91978.69	4840.864	190655.7	98666.98		Yes	94611.17	4791.176	196328.3	101717.2		Yes	97538.45	10838.83	348102.1	119162.4		Yes	104090.2	14357.02	348102.1	119162.4		Yes
520	87220.78	2236.43	183503.2	96282.43		Yes	88480.12	4708.427	188214.5	98154.43		Yes	91948.45	4658.28	193446.7	101501.3		Yes	94723.14	10233.53	348102.1	119162.4		Yes	97858.86	13978.85	348102.1	119162.4		Yes

f inert 0.063
435 142233.4 8503.183 248942.8 103609.2 Yes

Dummkopf Chart (Earth Escape)

Initial Mass vs. ISP, Payload = 94046 kg, Delta V = 3290 m/s



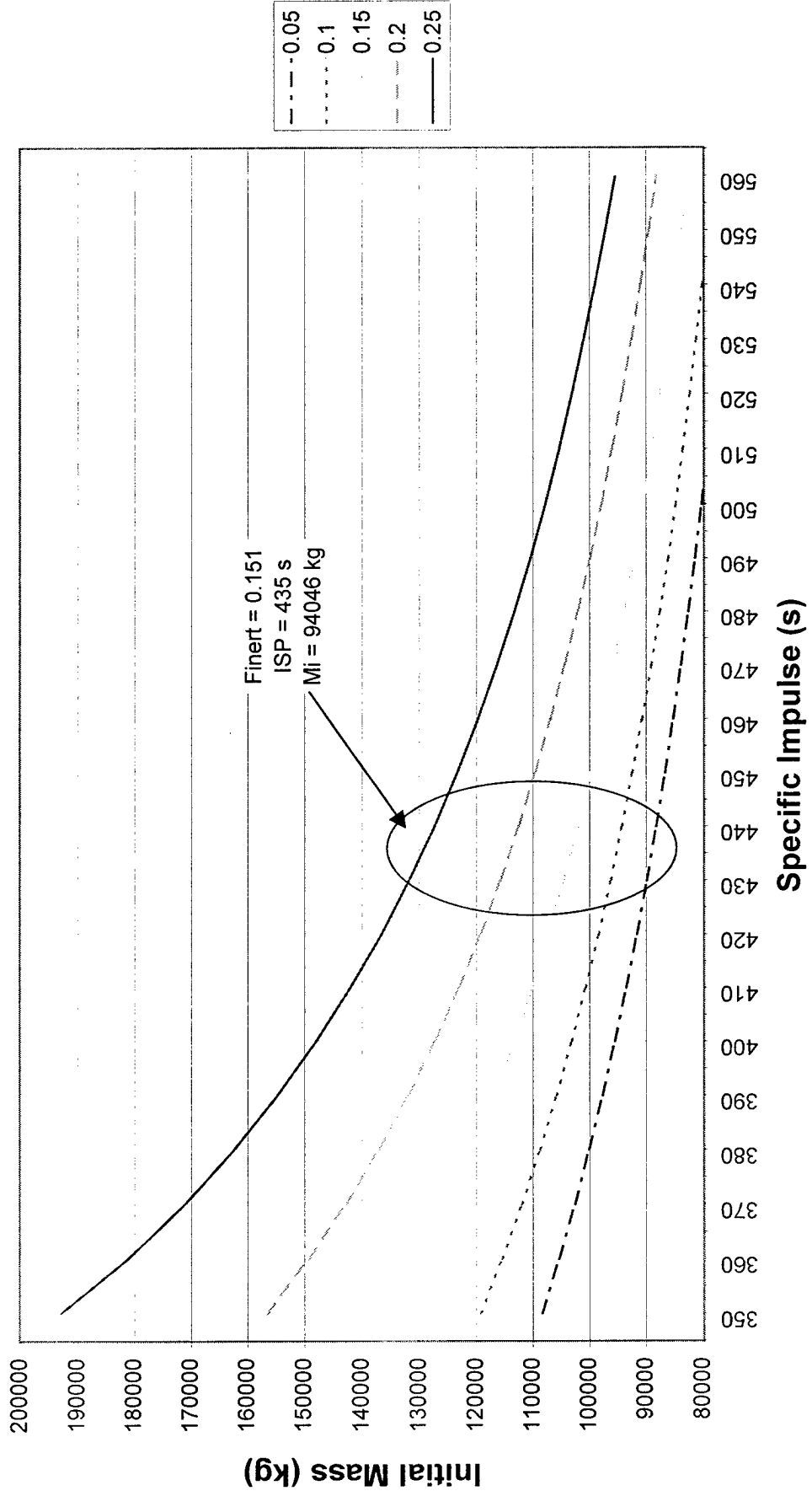
Information to determine the Downstop chart for Mars Insertion

M payload 41000 kg
dv 3087 m/s
gravity 9.81 m/s²

f ISP	0.05				0.1				0.15				0.2				0.25			
	Mprop	Minwet	MI	Min ISP	Feasible	Mprop	Minwet	MI	Min ISP	Feasible	Mprop	Minwet	MI	Min ISP	Feasible	Mprop	Minwet	MI	Min ISP	Feasible
350	64029.19	3368.957	100398.1	44369.96	104.3818	Yes	7064.48	7623.307	119293.9	48929.39	135.7779	Yes	15079.2	15079.2	15079.2	Yes	15079.2	15079.2	15079.2	Yes
360	64029.19	3368.957	100398.1	44369.96	104.3818	Yes	7064.48	7623.307	119293.9	48929.39	135.7779	Yes	15079.2	15079.2	15079.2	Yes	15079.2	15079.2	15079.2	Yes
370	55159.63	3598.822	102816.4	44400.82	Yes	6368.04	7096.469	111964.5	45096.45	Yes	13238.26	13238.26	13238.26	13238.26	Yes	13238.26	13238.26	13238.26	13238.26	Yes
380	56116.48	2952.948	100070	43953.5	Yes	6368.04	6777.849	108776.5	47777.65	Yes	71107.83	12548.44	12468.3	53548.44	Yes	61501.26	20275.31	142676.6	61375.31	Yes
390	53883.02	2835.948	87718.97	43935.95	Yes	59368.84	6485.44	105654.4	47485.44	Yes	64357.27	11357.17	11674.4	52923.99	Yes	76886.28	18221.57	137107.9	60221.57	Yes
400	51815.16	2727.113	95542.27	43727.11	Yes	55850.18	6216.897	103169.9	47216.89	Yes	61429.18	10940.44	11208.6	51840.44	Yes	72754.47	18188.62	131943.1	59188.62	Yes
410	48895.49	2625.078	93521.57	43526.08	Yes	53718.5	5968.722	100687.2	46898.72	Yes	58749.48	9367.25	10717.2	51847.55	Yes	69034.51	17558.63	127933.1	58258.63	Yes
420	46844.42	2444.339	89868.76	43444.34	Yes	51575.12	5655.48	97855.35	46575.35	Yes	55749.48	8367.25	10317.2	51847.55	Yes	65898.44	16551.1	125890.5	56652.11	Yes
430	44894.42	2362.339	86246.77	43362.34	Yes	47564.48	5328.278	94282.76	46378.28	Yes	54020.26	8532.897	10453.2	50532.89	Yes	62898.44	15751.1	123690.5	54652.11	Yes
440	43424.9	2285.521	88710.42	43285.52	Yes	46792.16	5143.573	92435.73	46143.57	Yes	51823.84	8163.031	102086.9	50163.03	Yes	59815.15	14953.78	115768.9	55653.78	Yes
450	42054.95	2213.419	85268.37	43213.42	Yes	44738.56	4970.951	90700.51	45970.95	Yes	48980.51	8020.081	99000.61	88920.09	Yes	54901.63	13725.41	109697.7	54725.41	Yes
460	40769.77	2145.617	83972.34	43145.62	Yes	43263.52	4809.28	88792.8	45809.28	Yes	48174.32	8501.35	97675.07	89501.35	Yes	52729.97	13182.49	106812.5	54182.48	Yes
470	39469.78	2071.498	81439.78	43021.49	Yes	41575.12	4655.48	86927.27	45615.93	Yes	46493.78	7937.016	93946.27	88927.27	Yes	50515.15	12453.78	103989.4	53153.78	Yes
480	38202.31	1910.648	78912.86	42910.65	Yes	40824.34	4514.927	86149.27	45514.93	Yes	43488.81	7687.437	92116.24	8687.44	Yes	48502.32	11750.58	97877.9	52375.58	Yes
490	37006.9	1834.1	82341	42834.1	Yes	38284.66	4253.851	83538.51	45253.85	Yes	40773.07	7195.248	89888.32	84195.25	Yes	43988.71	10987.43	89887.13	51897.43	Yes
500	35330.66	1859.572	78191.43	42699.57	Yes	37206.9	4134.1	82341	45134.1	Yes	38552.88	6979.822	87532.81	81979.82	Yes	42372.75	10643.19	84215.84	51843.19	Yes
510	33530.66	1859.572	78191.43	42699.57	Yes	35203.53	3913.982	80133.92	44813.98	Yes	36287.78	6403.728	83891.51	77565.12	Yes	40682.53	10325.18	81979.82	50510.12	Yes
520	31703.3	1721.227	75424.53	42721.23	Yes	34503.41	3811.49	78148.9	44813.98	Yes	36287.78	6403.728	83891.51	77565.12	Yes	40682.53	10325.18	81979.82	50510.12	Yes
530	30310.37	1675.499	74589.86	42679.48	Yes	33432.01	3714.658	76146.68	44714.67	Yes	35314.08	6231.887	82545.88	77231.8	Yes	37701.82	9425.458	88127.28	50053.48	Yes

Dummkopf Chart (Mars Insertion Orbit)

Initial Mass Vs. ISP, Payload = 41000 kg, Delta V = 3067 m/s



Appendix C: Top Level Mission Analysis

IMPRESS

T1000G

Cryogenic Storage System

Lee Gentile
Mars mission design

Given

dV	3290 m/s
Mass of Habitat Unit	30000 kg
Mass of Cryo Unit	11000 kg
Mass of Payload	94046.75849 kg
Max G (F/W)	6 earth G's
Thrust to Weight Ratio	1.2
g0	9.81 m/s ²

Thrust Requirement-top level analysis

ISP	435 sec	
finert	0.063	
mass prop	142234.5166 kg	add 20% for residual
max inert mass	9563.25992 kg	
max initial mass	245844.535 kg	
F	2894081.866 N	

Part 1: system mass and envelope

Me	3592.027051 kg
Le	15659.96208 cm
De	10346.35226 cm

Part 2: propellant

Oxygen and Hydrogen

O/F	3.8
ISP	435 sec

Part 3 & 4: Engine cycle and cooling approach

Expander cycle

Regenerative cooling

Part 5: Pressure levels

Exit pressure (Pe)	15000 Pa	selected
Chamber pressure (Pc)	11500000 Pa	selected
Atmospheric pressure(Pa)	650 Pa	vacuum
γ	1.23	O/F = 3.8
Tc	2800 K	O/F = 3.8
MW	9.8 kg/kmol	O/F = 3.8
c*	2402.69663 m/s	1.02 c* eff.

Output

Me	4.627484385 eq 3.100
ϵ	52.52696338 to get Pe
mdot propellant	678.1918207 eq 1.7
At	0.141694713 eq 3.133
Ae	7.442793014 using ϵ
mdot ox	536.901858 O/F = 3.8
mdot fuel	141.2899626 O/F = 3.8

Determination of Propellant Needed at Mars

Mpayload	41000 kg
Minert	9380.228 kg
Isp	435 sec
g0	9.81 m/s ²
Delta V	3067 m/s
f inert	0.151 selected

M prop	53046.76 kg
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f inert	0.150259 calculated
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c*	2402.697
gamma	1.23
g0	9.81
Pe	15000
Pa	650
lambda	0.95

Pc	Me	ϵ	ISP
0	#DIV/0!	#DIV/0!	#DIV/0!
500000	2.83838	4.991658	378.6091
1000000	3.220975	8.245424	394.7603
1500000	3.446272	11.12957	403.0735
2000000	3.607461	13.80014	408.5232
2500000	3.733469	16.32383	412.5138
3000000	3.837156	18.73714	415.6299
3500000	3.925385	21.06289	418.168
4000000	4.002261	23.31658	420.2979
4500000	4.070435	25.50935	422.1254
5000000	4.131721	27.64958	423.7208
5500000	4.187416	29.74377	425.1327
6000000	4.238481	31.79709	426.3964
6500000	4.285645	33.81375	427.538
7000000	4.329479	35.79724	428.5775
7500000	4.370433	37.75048	429.5304
8000000	4.408874	39.67596	430.4091
8500000	4.445101	41.57582	431.2235
9000000	4.479361	43.45192	431.9816
9500000	4.511865	45.30589	432.6903
10000000	4.542788	47.13918	433.355
10500000	4.572281	48.95304	433.9806
11000000	4.600476	50.74863	434.5711
11500000	4.627484	52.52696	435.1298
12000000	4.653406	54.28897	435.6599
12500000	4.678327	56.03549	436.1638
13000000	4.702325	57.76728	436.6439
13500000	4.725467	59.48504	437.102
14000000	4.747815	61.18941	437.5401
14500000	4.769423	62.88098	437.9596
15000000	4.79034	64.5603	438.362
15500000	4.81061	66.22786	438.7484
16000000	4.830274	67.88413	439.12
16500000	4.849367	69.52955	439.4778
17000000	4.867924	71.16452	439.8227
17500000	4.885973	72.78943	440.1555
18000000	4.903545	74.40461	440.477
18500000	4.920663	76.01041	440.7879
19000000	4.937351	77.60713	441.0888
19500000	4.953632	79.19508	441.3802
20000000	4.969526	80.77452	441.6628
20500000	4.98505	82.34573	441.9369
21000000	5.000222	83.90893	442.2031
21500000	5.015059	85.46438	442.4617
22000000	5.029576	87.01229	442.7131
22500000	5.043786	88.55287	442.9577
23000000	5.057703	90.08632	443.1958
23500000	5.071339	91.61283	443.4278
24000000	5.084705	93.13259	443.6538
24500000	5.097813	94.64576	443.8742
25000000	5.110672	96.15251	444.0893

Estimate Masses and size tanks

Isp 435 sec
 f inert 0.063
 payload 94046.76 kg
 dV 3290 m/s

Estimate Masses

Mprop 142234.5 eq. 1.27 20% added for blow down extra
 Minert 9563.26 eq. 1.24
 Mfinal 103610
 Minitial 245844.5
 Mfuel 29632.19 O/F = 3.8
 Mox 112602.3 O/F = 3.8
 Vfuel 417.3548 71
 Vox 98.60099 1142

Size Tanks

	spherical tanks	
	Composite Ox	Composite Fuel
Density Prop.	1142	71
Mass Prop/Press.	112602.3256	29632.19096
MEOP tank (Pa)	2000000	2000000
Burst Press. (Pa)	4000000	4000000
Ftu (Pa)	1.16E+09	1.16E+09
Tank Density (kg/m ³)	1550	1550
Vol. (5% Ullage)	103.531035	438.2225423
radius (m)	2.91291167	4.711964772
area (m ²)	106.6263382	279.0062512
wall thickness (m)	0.005022261	0.008124077
mass tank (kg)	830.0332974	3513.335899
tank factor	50800	50800
mass tank using TF	830.9938834	3517.401834

Thrust Chamber

N2O4/RP-1

Input

Pc (Pa)	11500000
c*	2402.69663
gamma	1.23
At	0.141694713
Ae	7.442793014
L* Table 5.6	0.5
Mc	0.4
Ftu	3.10E+08 columbium
mult. factor	3
cham cont angle	45
dens comb cham	8500 columbium
expansion ratio	52.52696338

Output

Ac (m^2)	0.329392156
Chamber Len (m)	0.215085136
Chamber Dia (m)	0.647607225
Troat Dia (m)	0.424748528
Len:Dia ratio	0.332122817
Cham Thick (m)	0.036036209
Mass Cham (kg)	169.5331742

Nozzle**Input**

Nozzle Throat Dia	0.424749
Ae	7.442793
Nozzle exit Diam.	3.078386
Cone 1/2 angle (deg)	20
Lf for eff=.985 fig 5.25	0.6
thick. throat wall (m)	0.018018
Using 1/2 chamber wall	
thick. nozzle exit	0.009009
density nozzle mater.	8500 columbium

Output

Ln (m) 15 deg nozzle	3.645404
Ln bell nozzle (m)	2.187242
theta n	38
theta e	13

Using a non-tapered bell nozzle

mass nozzle (kg)	1843.324
------------------	----------

Using a tapered nozzle

f1	-0.004119
f2	0.606617
mass nozzle (kg)	1266.132

Mass Summary

Total engine mass	2871.331097
(using 50% for chamber & nozzle)	
Mass injectors (15%)	717.8327743
mass oxidizer tank	830.9938834
mass fuel tank	3517.401834
support structure	721.9726815
(10% of tank & engine masses)	
feed system	720.695954
(system level est.-engine mass)	
Total Inert mass	9380.228224
propellant mass	142234.5166
payload mass	94046.75849
Final mass	103426.9867
Inert mass fraction	0.061868839
Initial mass of vehicle	245661.5033
Thrust	2894081.866
F/W	1.200894068
Check G limit	2.852383613

Appendix D: Top Level Analysis Using Different Technology Combinations

IMPRESS, T1000G without cryogenics

IMPRESS, Cryogenics with Titanium tanks

Cryogenics, T1000G tanks without IMPRESS

Cryogenics, Titanium tanks without IMPRESS

Lee Gentile
Mars mission design
IMPRESS, T1000G tanks without Cryogenic Storage

Given

dV	3290 m/s
Mass of Habitat Unit	30000 kg
Mass of Cryo Unit	11000 kg
Mass of Payload	4590614.889 kg
Max G (FW)	6 earth G's
Thrust to Weight Ratio	1.2
g0	9.81 m/s ²

Thrust Requirement-top level analysis

ISP	435 sec	
finert	0.347	
mass prop	16728974.15 kg	add 20% for residual
max inert mass	8889669.264 kg	
max initial mass	30209258.3 kg	
F	355623388.7 N	

Part 1: system mass and envelope

Me	268955.1642 kg
Le	11145.76349 cm
De	8570.45574 cm

Part 2: propellant

Oxygen and Hydrogen

O/F	3.8
ISP	435 sec

Part 3 & 4: Engine cycle and cooling approach

Expander cycle
Regenerative cooling

Part 5: Pressure levels

Exit pressure (Pe)	15000 Pa	selected
Chamber pressure (Pc)	11500000 Pa	selected
Atmospheric pressure(Pa)	650 Pa	vacuum
γ	1.23	O/F = 3.8
Tc	2800 K	O/F = 3.8
MW	9.8 kg/kmol	O/F = 3.8
c*	2402.69663 m/s	1.02 c* eff.

Output

Me	4.627484385 eq 3.100
ϵ	52.52696338 to get Pe
mdot propellant	83335.88497 eq 1.7
At	17.41137826 eq 3.133
Ae	914.5668284 using ϵ
mdot ox	65974.24227 O/F = 3.8
mdot fuel	17361.6427 O/F = 3.8

Determination of Propellant Needed at Mars

Mpayload	41000 kg
Minert	9784866 kg
Isp	435 sec
g0	9.81 m/s^2
Delta V	3067 m/s
f inert	0.485 selected

M prop	4549615 kg
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f inert	0.68261 calculated
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Estimate Masses and size tanks

Isp 435 sec
f inert 0.347
payload 4590615 kg
dV 3290 m/s

Estimate Masses

Mprop 16728974 eq. 1.27 20% added for blow down extra
Minert 8889669 eq. 1.24
Mfinal 13480284
Minitial 30209258
Mfuel 3485203 O/F = 3.8
Mox 13243771 O/F = 3.8
Vfuel 719171.6 4.846135207
Vox 172169 76.92307692

Size Tanks

	spherical tanks	
	Composite Ox	Composite Fuel
Density Prop.	76.92307692	4.846135207
Mass Prop/Press.	13243771.2	3485202.948
MEOP tank (Pa)	2000000	2000000
Burst Press. (Pa)	4000000	4000000
Ftu (Pa)	1.16E+09	1.16E+09
Tank Density (kg/m^3)	1550	1550
Vol. (5% Ullage)	180777.4769	755130.2097
radius (m)	35.07668724	56.49080262
area (m^2)	15461.33553	40101.93738
wall thickness (m)	0.060477047	0.097397936
mass tank (kg)	1449336.668	6054061.164
tank factor	50800	50800
mass tank using TF	1451013.965	6061067.444

Density

Temp Init (K)
Press Init
Press Fin
Temp Fin (K)
R
Density init
Density fin

Oxygen

100
2000000
50000
2.5
260
76.92307692
1.923076923

Hydrogen

100
2000000
50000
2.5
4127
4.846135207
0.12115338

Thrust Chamber

N2O4/RP-1

Input

Pc (Pa)	11500000
c*	2402.69663
gamma	1.23
At	17.41137826
Ae	914.5668284
L* Table 5.6	0.5
Mc	0.4
Ftu	3.10E+08 columbium
mult. factor	3
cham cont angle	45
dens comb cham	8500 columbium
expansion ratio	52.52696338

Output

Ac (m^2)	40.47554979
Chamber Len (m)	0.215085136
Chamber Dia (m)	7.178793115
Troat Dia (m)	4.708381392
Len:Dia ratio	0.029961183
Cham Thick (m)	0.399465101
Mass Cham (kg)	64818.86625

Nozzle**Input**

Nozzle Throat Dia	4.708381
Ae	914.5668
Nozzle exit Diam.	34.12422
Cone 1/2 angle (deg)	20
Lf for eff=.985 fig 5.25	0.6
thick. throat wall (m)	0.199733
Using 1/2 chamber wall	
thick. nozzle exit	0.099866
density nozzle mater.	8500 columbium

Output

Ln (m) 15 deg nozzle	40.40968
Ln bell nozzle (m)	24.24581
theta n	38
theta e	13

Using a non-tapered bell nozzle

mass nozzle (kg)	2510851
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Using a tapered nozzle

f1	-0.004119
f2	0.606617
mass nozzle (kg)	1724640

Mass Summary

Total engine mass	3578917.058
(using 50% for chamber & nozzle)	
Mass injectors (15%)	894729.2645
mass oxidizer tank	1451013.965
mass fuel tank	6061067.444
support structure	1109099.847
(10% of tank & engine masses)	
feed system	-3309961.894
(system level est.-engine mass)	
Total Inert mass	9784865.685
propellant mass	16728974.15
payload mass	4590614.889
Final mass	14375480.57
Inert mass fraction	0.369047477
Initial mass of vehicle	31104454.72
Thrust	355623388.7
F/W	1.165463606
Check G limit	2.521732041

Lee Gentile
Mars mission design
IMPRESS, Cryogenic Storage with Titanium Tanks

Given

dV	3290 m/s
Mass of Habitat Unit	30000 kg
Mass of Cryo Unit	11000 kg
Mass of Payload	289417.0399 kg
Max G (FW)	6 earth G's
Thrust to Weight Ratio	1.2
g0	9.81 m/s ²

Thrust Requirement-top level analysis

ISP	435 sec	
finert	0.24	
mass prop	637368.6101 kg	add 20% for residual
max inert mass	201274.2979 kg	
max initial mass	1128059.948 kg	
F	13279521.71 N	

Part 1: system mass and envelope

Me	13700.55093 kg
Le	731.6630503 cm
De	494.5639171 cm

Part 2: propellant

Oxygen and Hydrogen

O/F	3.8
ISP	435 sec

Part 3 & 4: Engine cycle and cooling approach

Expander cycle
Regenerative cooling

Part 5: Pressure levels

Exit pressure (Pe)	15000 Pa	selected
Chamber pressure (Pc)	11500000 Pa	selected
Atmospheric pressure(Pa)	650 Pa	vacuum
γ	1.23	O/F = 3.8
Tc	2800 K	O/F = 3.8
MW	9.8 kg/kmol	O/F = 3.8
c*	2402.69663 m/s	1.02 c* eff.

Output

Me	4.627484385 eq 3.100
ϵ	52.52696338 to get Pe
mdot propellant	3111.889511 eq 1.7
At	0.650167517 eq 3.133
Ae	34.15132535 using ϵ
mdot ox	2463.579197 O/F = 3.8
mdot fuel	648.3103149 O/F = 3.8

Determination of Propellant Needed at Mars

Mpayload	41000 kg
Minert	194560.3 kg
Isp	435 sec
g0	9.81 m/s^2
Delta V	3067 m/s
f inert	0.44 selected

M prop	248417 kg
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f inert	0.43921 calculated
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Estimate Masses and size tanks

Isp 435 sec
 f inert 0.24
 payload 289417 kg
 dV 3290 m/s

Estimate Masses

Mprop 637368.6 eq. 1.27 20% added for blow down extra
 Minert 201274.3 eq. 1.24
 Mfinal 490691.3
 Minitial 1128060
 Mfuel 132785.1 O/F = 3.8
 Mox 504583.5 O/F = 3.8
 Vfuel 1870.213 71
 Vox 441.8419 1142

Size Tanks

	spherical tanks	
	Composite Ox	Composite Fuel
Density Prop.	1142	71
Mass Prop/Press.	504583.483	132785.1271
MEOP tank (Pa)	2000000	2000000
Burst Press. (Pa)	4000000	4000000
Ftu (Pa)	1.23E+09	1.23E+09
Tank Density (kg/m ³)	4460	4460
Vol. (5% Ullage)	463.9340255	1963.723711
radius (m)	4.802372816	7.768382327
area (m ²)	289.8154995	758.3523679
wall thickness (m)	0.007808736	0.012631516
mass tank (kg)	10093.39392	42722.96463
tank factor	6350	6350
mass tank using TF	29790.20447	126094.9352

Thrust Chamber

N2O4/RP-1

Input

Pc (Pa)	11500000
c*	2402.69663
gamma	1.23
At	0.650167517
Ae	34.15132535
L* Table 5.6	0.5
Mc	0.4
Ftu	3.10E+08 columbium
mult. factor	3
cham cont angle	45
dens comb cham	8500 columbium
expansion ratio	52.52696338

Output

Ac (m ²)	1.511418987
Chamber Len (m)	0.215085136
Chamber Dia (m)	1.387226882
Troat Dia (m)	0.909845588
Len:Dia ratio	0.155046834
Cham Thick (m)	0.077192464
Mass Cham (kg)	963.9118645

Nozzle**Input**

Nozzle Throat Dia	0.909846
Ae	34.15133
Nozzle exit Diam.	6.59415
Cone 1/2 angle (deg)	20
Lf for eff=.985 fig 5.25	0.6
thick. throat wall (m)	0.038596
Using 1/2 chamber wall	
thick. nozzle exit	0.019298
density nozzle mater.	8500 columbium

Output

Ln (m) 15 deg nozzle	7.808749
Ln bell nozzle (m)	4.68525
theta n	38
theta e	13

Using a non-tapered bell nozzle

mass nozzle (kg)	18117.95
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Using a tapered nozzle

f1	-0.004119
f2	0.606617
mass nozzle (kg)	12444.76

Mass Summary

Total engine mass	26817.34486
(using 50% for chamber & nozzle)	
Mass injectors (15%)	6704.336216
mass oxidizer tank	29790.20447
mass fuel tank	126094.9352
support structure	18270.24845
(10% of tank & engine masses)	
feed system	-13116.79393
(system level est.-engine mass)	
Total Inert mass	194560.2752
propellant mass	637368.6101
payload mass	289417.0399
Final mass	483977.3151
Inert mass fraction	0.233866474
Initial mass of vehicle	1121345.925
Thrust	13279521.71
F/W	1.207184961
Check G limit	2.796973939

Lee Gentile
Mars mission design
Cryogenic Storage with T1000G tanks without IMPRESS

Given

dV	6357 m/s
Mass of Habitat Unit	30000 kg
Mass of Cryo Unit	11000 kg
Mass of Payload	41000 kg
Max G (F/W)	6 earth G's
Thrust to Weight Ratio	1.2
g0	9.81 m/s^2

Thrust Requirement-top level analysis

ISP	435 sec	
finert	0.054	
mass prop	210274.5125 kg	add 20% for residual
max inert mass	12002.98486 kg	
max initial mass	263277.4974 kg	
F	3099302.699 N	

Part 1: system mass and envelope

Me	3811.939654 kg
Le	421.9807881 cm
De	254.4125507 cm

Part 2: propellant

Oxygen and Hydrogen

O/F	3.8
ISP	435 sec

Part 3 & 4: Engine cycle and cooling approach

Expander cycle
Regenerative cooling

Part 5: Pressure levels

Exit pressure (Pe)	15000 Pa	selected
Chamber pressure (Pc)	11500000 Pa	selected
Atmospheric pressure(Pa)	650 Pa	vacuum
γ	1.23	O/F = 3.8
Tc	2800 K	O/F = 3.8
MW	9.8 kg/kmol	O/F = 3.8
c*	2402.69663 m/s	1.02 c* eff.

Output

Me	4.627484385 eq 3.100
ϵ	52.52696338 to get Pe
mdot propellant	726.2827515 eq 1.7
At	0.151742358 eq 3.133
Ae	7.970565295 using ϵ
mdot ox	574.9738449 O/F = 3.8
mdot fuel	151.3089066 O/F = 3.8

Estimate Masses and size tanks

Isp 435 sec
 f inert 0.054
 payload 41000 kg
 dV 6357 m/s

Estimate Masses

Mprop 210274.5 eq. 1.27 20% added for blow down extra
 Minert 12002.98 eq. 1.24
 Mfinal 53002.98
 Minitial 263277.5
 Mfuel 43807.19 O/F = 3.8
 Mox 166467.3 O/F = 3.8
 Vfuel 617.0027 71
 Vox 145.7682 1142

Size Tanks

	spherical tanks	
	Composite Ox	Composite Fuel
Density Prop.	1142	71
Mass Prop/Press.	166467.3224	43807.19011
MEOP tank (Pa)	2000000	2000000
Burst Press. (Pa)	4000000	4000000
Ftu (Pa)	1.16E+09	1.16E+09
Tank Density (kg/m ³)	1550	1550
Vol. (5% Ullage)	153.056645	647.8528115
radius (m)	3.318342378	5.367794894
area (m ²)	138.3732848	362.0776265
wall thickness (m)	0.00572128	0.009254819
mass tank (kg)	1227.092067	5193.992368
tank factor	50800	50800
mass tank using TF	1228.512164	5200.003303

Thrust Chamber

N2O4/RP-1

Input

Pc (Pa)	11500000
c*	2402.69663
gamma	1.23
At	0.151742358
Ae	7.970565295
L* Table 5.6	0.5
Mc	0.4
Ftu	3.10E+08 columbium
mult. factor	3
cham cont angle	45
dens comb cham	8500 columbium
expansion ratio	52.52696338

Output

Ac (m^2)	0.352749523
Chamber Len (m)	0.215085136
Chamber Dia (m)	0.670175083
Troat Dia (m)	0.439550192
Len:Dia ratio	0.320938724
Cham Thick (m)	0.037292001
Mass Cham (kg)	182.8794854

Nozzle**Input**

Nozzle Throat Dia	0.43955
Ae	7.970565
Nozzle exit Diam.	3.185661
Cone 1/2 angle (deg)	20
Lf for eff=.985 fig 5.25	0.6
thick. throat wall (m)	0.018646
Using 1/2 chamber wall	
thick. nozzle exit	0.009323
density nozzle mater.	8500 columbium

Output

Ln (m) 15 deg nozzle	3.772439
Ln bell nozzle (m)	2.263464
theta n	38
theta e	13

Using a non-tapered bell nozzle

mass nozzle (kg)	2042.826
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Using a tapered nozzle

f1	-0.004119
f2	0.606617
mass nozzle (kg)	1403.165

Mass Summary

Total engine mass	3172.089902
(using 50% for chamber & nozzle)	
Mass injectors (15%)	793.0224755
mass oxidizer tank	1228.512164
mass fuel tank	5200.003303

support structure	960.0605369
(10% of tank & engine masses)	
feed system	639.8497522
(system level est.-engine mass)	

Total Inert mass	11993.53813
propellant mass	210274.5125
payload mass	41000
Final mass	52993.53813

Inert mass fraction	0.053959794
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Initial mass of vehicle	263268.0507
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Thrust	3099302.699
F/W	1.200043059

Check G limit	5.961726807
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Lee Gentile
Mars mission design
Cryogenic Storage with Titanium Tanks without IMPRESS

Given

dV	6357 m/s
Mass of Habitat Unit	30000 kg
Mass of Cryo Unit	11000 kg
Mass of Payload	41000 kg
Max G (F/W)	6 earth G's
Thrust to Weight Ratio	1.2
g0	9.81 m/s ²

Thrust Requirement-top level analysis

ISP	435 sec	
finert	0.225	
mass prop	66511024.08 kg	add 20% for residual
max inert mass	19309652.15 kg	
max initial mass	85861676.23 kg	
F	1010763653 N	

Part 1: system mass and envelope

Me	704664.2419 kg
Le	31075.13031 cm
De	24025.21456 cm

Part 2: propellant

Oxygen and Hydrogen

O/F	3.8
ISP	435 sec

Part 3 & 4: Engine cycle and cooling approach

Expander cycle
Regenerative cooling

Part 5: Pressure levels

Exit pressure (Pe)	15000 Pa	selected
Chamber pressure (Pc)	11500000 Pa	selected
Atmospheric pressure(Pa)	650 Pa	vacuum
γ	1.23	O/F = 3.8
Tc	2800 K	O/F = 3.8
MW	9.8 kg/kmol	O/F = 3.8
c*	2402.69663 m/s	1.02 c* eff.

Output

Me	4.627484385 eq 3.100
ϵ	52.52696338 to get Pe
mdot propellant	236859.7965 eq 1.7
At	49.48715086 eq 3.133
Ae	2599.409761 using ϵ
mdot ox	187514.0056 O/F = 3.8
mdot fuel	49345.79094 O/F = 3.8

Estimate Masses and size tanks

Isp 435 sec
 f inert 0.225
 payload 41000 kg
 dV 6357 m/s

Estimate Masses

Mprop 66511024 eq. 1.27 20% added for blow down extra
 Minert 19309652 eq. 1.24
 Mfinal 19350652
 Minitial 85861676
 Mfuel 13856463 O/F = 3.8
 Mox 52654561 O/F = 3.8
 Vfuel 195161.5 71
 Vox 46107.32 1142

Size Tanks

	spherical tanks	
	Composite Ox	Composite Fuel
Density Prop.	1142	71
Mass Prop/Press.	52654560.73	13856463.35
MEOP tank (Pa)	2000000	2000000
Burst Press. (Pa)	4000000	4000000
Ftu (Pa)	1.23E+09	1.23E+09
Tank Density (kg/m^3)	4460	4460
Vol. (5% Ullage)	48412.68719	204919.5284
radius (m)	22.60946018	36.57336437
area (m^2)	6423.773961	16808.91533
wall thickness (m)	0.03676335	0.059468885
mass tank (kg)	1053271.146	4458249.252
tank factor	6350	6350
mass tank using TF	3108683.069	13158324.92

Thrust Chamber

N2O4/RP-1

Input

Pc (Pa)	11500000
c*	2402.69663
gamma	1.23
At	49.48715086
Ae	2599.409761
L* Table 5.6	0.5
Mc	0.4
Ftu	3.10E+08 columbium
mult. factor	3
cham cont angle	45
dens comb cham	8500 columbium
expansion ratio	52.52696338

Output

Ac (m^2)	115.0408433
Chamber Len (m)	0.215085136
Chamber Dia (m)	12.1026671
Troat Dia (m)	7.937820698
Len:Dia ratio	0.017771714
Cham Thick (m)	0.673454863
Mass Cham (kg)	278483.2681

Nozzle**Input**

Nozzle Throat Dia	7.937821
Ae	2599.41
Nozzle exit Diam.	57.52974
Cone 1/2 angle (deg)	20
Lf for eff=.985 fig 5.25	0.6
thick. throat wall (m)	0.336727
Using 1/2 chamber wall	
thick. nozzle exit	0.168364
density nozzle mater.	8500 columbium

Output

Ln (m) 15 deg nozzle	68.12634
Ln bell nozzle (m)	40.87581
theta n	38
theta e	13

Using a non-tapered bell nozzle

mass nozzle (kg)	12031224
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Using a tapered nozzle

f1	-0.004119
f2	0.606617
mass nozzle (kg)	8263943

Mass Summary

Total engine mass	17084852.8
(using 50% for chamber & nozzle)	
Mass injectors (15%)	4271213.199
mass oxidizer tank	3108683.069
mass fuel tank	13158324.92
support structure	3335186.079
(10% of tank & engine masses)	
feed system	-16380188.55
(system level est.-engine mass)	
Total Inert mass	24578071.51
propellant mass	66511024.08
payload mass	41000
Final mass	24619071.51
Inert mass fraction	0.26982452
Initial mass of vehicle	91130095.59
Thrust	1010763653
F/W	1.130625517
Check G limit	4.185129867